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ADVANCED DESIGN PROGRAM
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UNIVERSITY SPONSOR
BOEING COMMERCIAL AIRPLANE COMPANY

FINAL DESIGN PROPOSAL

ALPHA GROUP - THE BEHEMOTH APTERYX

A Proposal in Response to a Commercial Air
Transportation Study

May 1991

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EXECUTIVE SUMMARY

Alpha Design Group has formulated a design for an aircraft called *The Behemoth Apteryx.* The design is a compilation of efforts both to fulfill requirements imposed by the project definition and to optimize efficiency in both performance and construction. The following are the basic characteristics of *The Behemoth Apteryx.*

The first constraint we wanted to fulfill was a wingspan limited to five feet in order to be able to utilize all gates in Aeroworld while having a solid, unhinged wing. Our calculations led us to choose a SPICA airfoil section with a wingspan of 60 in. and a chord of 14 in. This put us in a precarious position of flying relatively close to C_{Lmax}, Mach 1 cruise velocity, and α_{stall}. We recognized these risks and decided that they could be overcome in our efforts to satisfy our self imposed requirements.

With such short wingspan and thus small area and Aspect Ratio, the next critical constraint was minimal weight. The small area meant a large wing loading, thus every effort was made to minimize weight.

Considering our two major limiting factors, the consequential design can be summarized as follows: Propulsion to be provided by an Astro-15 electric motor and a 650 mah battery pack. The fuselage is 44 in. long with a maximum width of 7 in. and will hold 50 passengers + 2 crew members. The structure will consist of a balsa wood and spruce truss structure for the fuselage and balsa wood spars and ribs for the wing. The entire aircraft will be covered with Monokote shrinkable plastic coating. Control will be done by means of an elevator, a rudder, and ailerons. For our recommended market, fleet size, and ticket price, the purchasing airline could make $840 million (before taxes) per year and Alpha Design would make $4,316,800 on the sale of that fleet.

Potential problems with *The Behemoth Apteryx* result mostly from our five foot wingspan restriction. In order to maintain stable and comfortable flight, we must cruise at 32 ft./sec or M=.91. The take-off speed is 29 ft./sec which is also relatively high. However, our design is very versatile in that it can access any airport gate and any runway without any additional ground crew handling associated with a hinged wing. It also is extremely easy and inexpensive to build which keeps the purchase price down, thus making it a very marketable aircraft. With our plane, we can beat all existing modes of travel in cost, speed and convenience. This would make air transportation the ultimate in travel in Aeroworld. We feel that the benefits we receive from our self imposed restrictions well justify the risks in design, and these benefits have thus driven our design.

16"

2"

60"

6"

ELEVATOR
24" × 1.5"

7"

AILERON
18" × 1.25"

14"

6"

4"

RUDDER
10" × 2"

6"

48"
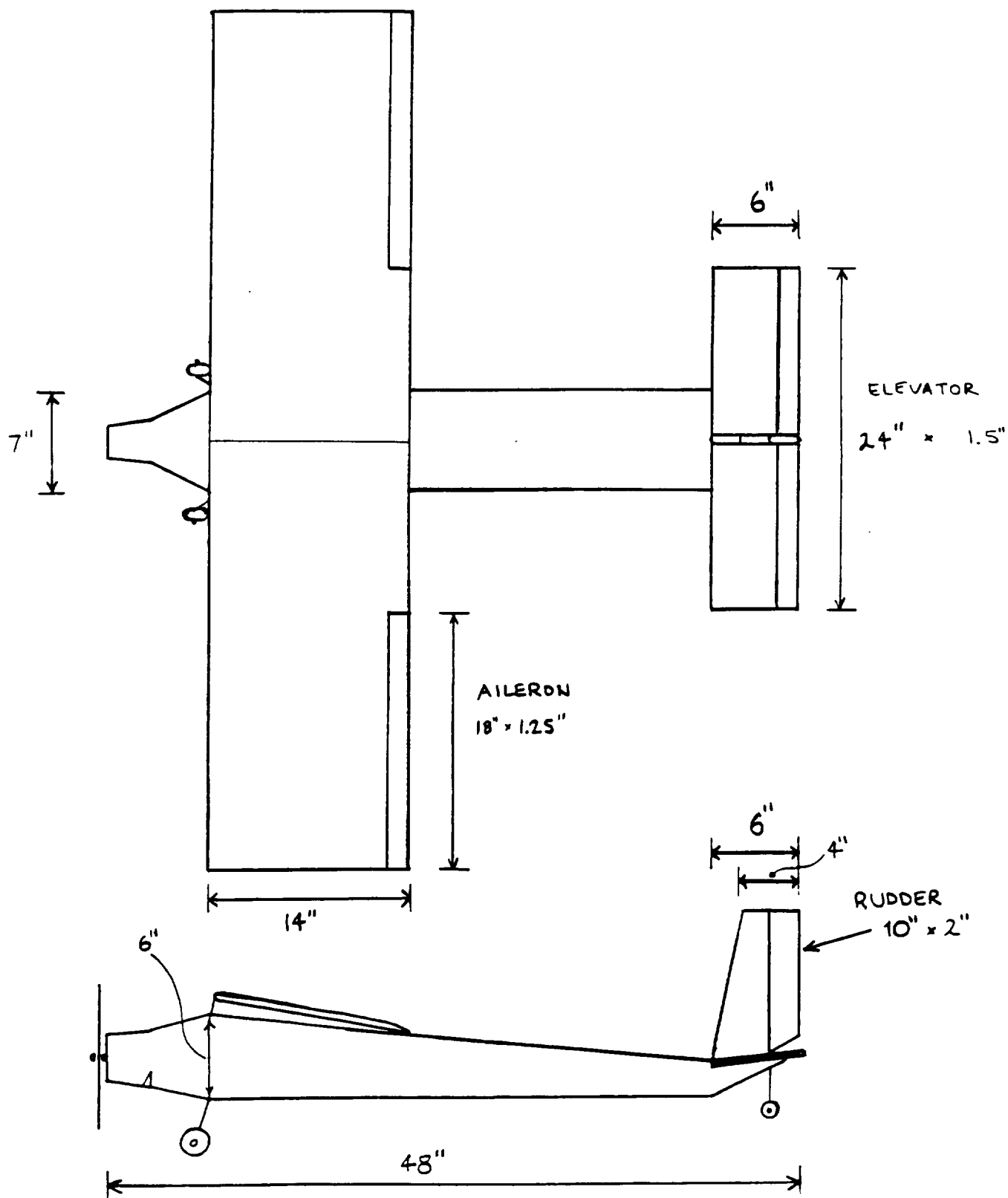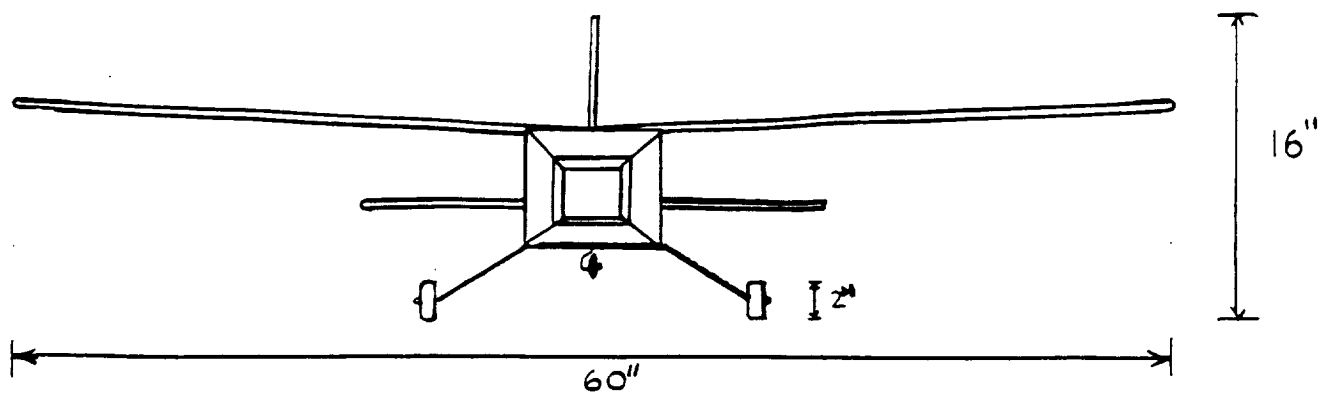
NOMENCLATURE

RFP...Request For Proposal

C_l...Lift Coefficient for infinite wing

i_w...wing mounted incidence angle

C_D...Drag Coefficient

C_{D_0}...Induced Drag Coefficient

AR...Aspect Ratio

e...Oswald efficiency factor

C_{D_π}...C_D for section "π"

A_π...Area of section "π"

S_{ref}...Reference area

$\dfrac{L}{D}$...Lift to Drag ratio

α...Angle of attack

C_m...Moment Coefficient

C_{m_0}...Moment Coefficient at $\alpha=0$

C_{m_α}...Slope of C_m vs. α

c.g. ....Center of Gravity

Δx...length of fuselage increments

W_f...average width of fuselage sections

x_i distance from c.g.

$\bar{x}$...x for wing section

$\dfrac{x_i}{Cr_e}$...x/ref chord length

$\dfrac{x_i}{l_h}$...x/tail moment arm

$\dfrac{\partial \varepsilon_u}{\partial \alpha}$...change in downwash with α

$C_{m_{\alpha f}}$...contribution to C_{m_α} of fuselage

$C_{m_{0_t}}$...contribution to C_{m_0} of tail

V_H ....Horizontal tail volume ratio

$C_{L_{\alpha_t}}$...Lift Slope for tail surface

ε_0...downwash at $\alpha=0$

i_t...tail mounted incidence angle

c...chord length

i_f ....fuselage incidence angle

$C_{m_{o_f}}$ ....contribution to C_{m_o} of fuselage

C_{N_β} ....Yaw moment coefficient due to sideslip

Se/St...ratio of elevator are to tail area

$C_{m_{\delta e}}$ ....change in C_m due to elevator deflection

$C_{l_{\delta a}}$ ....change in C_l due to aileron deflection

C_t ....Thrust Coefficient

C_p ....Power Coefficient

J ....Advance Ratio

DR&O..Design Requirements and Objectives

E ....Young's Modulus

σ_{xx} ....stress

g's ....units of gravity force

n ....load factor

X_{gr} ....ground roll distance

W ....Weight

S ....wing planform area

μ_r ....rolling friction coefficient

RPV ....Remotely Piloted Vehicle

mah ....milli-amp hours

γ ....glide angle

RWT...Real World Time

AWT...Aero World Time
 (1min RWT=30min AWT)

DATA SUMMARY

GENERAL:
Weight = 61.0 oz
50 passenger capacity
8000 foot range with redirect and loiter time
Take off and Landing Ground Roll ≤ 50 feet
Optimum Turning Radius ≤ 60 feet
Cruise Speed = 32 fps
Cruise Altitude = 20 feet
Optimum Maximum Endurance = 8.2 minutes (@ 25 fps)
Optimum Maximum Range = 19705.7 feet (@ 44 fps)
Optimum Cruise Endurance = 8.5 minutes (@ 32 fps)
Optimum Cruise Range = 16236.8 feet (@ 32 fps)
Power Required at Cruise = 15.043 Watts (@ i = 4.17 amps)
L/D_{max} = 10.8
Glide Angle, γ = 5.28°
Best Glide Range, $X_{best\ range}$ = 270.75 ft

PROPULSION SYSTEM:
Motor: ASTRO-15
Propeller: Topflight 10-4
Power: Gates 650SCR batteries (12)

CONTROLS:
Ailerons: 18 in. x 1.25 in.
Elevator: 24.5 in. x 1.25 in.
Rudder: 10 in. x1.5 in.
3 servos plus throttle control and linkages
Servo battery pack

GENERAL CONFIGURATION:
Fuselage: 44 in. L x 7 in. W x 5 in. max H
Wing: Span = 60 in. Chord= 14 in. no taper, sweep or twist
Horizontal Tail: 24.5 in. x 6.5 in.
Vertical Tail: 10 in. H x 6 in. (root) x 4 in. (tip)
Landing Gear: Tail Dragger configuration--2 in. front wheels, 1 in. rear wheel
Zero Lift Drag Coefficient, C_{do} = .04
Wing Oswald Efficiency Factor: e = .76

1.0 MISSION SCOPING

1.1 Mission Definition

1.1.1 Request for Proposal

The following is a copy of the Request for Proposal:

Commercial Air Transportation System Design

Commercial transports operate on a wide variety of missions ranging from short 20 minute commuter hops to extended 14 hour flights which travel across oceans and continents. In order to satisfy this wide range of mission requirements "families" of aircraft have been developed. Each basic airplane in the family was initially designed for a specific purpose but from that basic aircraft numerous derivative aircraft are often developed. The design of the basic aircraft must be sensitive to the fact that derivative aircraft can be developed.

Though they may differ in size and performance, all commercial designs must also possess one common denominator; they must be able to generate a profit, which requires compromises between technology and economics. The objective of this project will be to gain some insight into the problems and trade-offs involved in the design of a commercial transport system. This project will simulate numerous aspects of the overall systems design process so that you will be exposed to many of the conflicting requirements encountered in a systems design. In order to do so in the limited time allowed for this single course, a "hypothetical" world has been developed and you will be provided with information on geography, demographics and economic factors. The project is formulated in such a fashion that you will be asked to perform a systems design study but will provide an opportunity to identify those factors which have the most significant influence on the system design and design process. Formulating the project in this manner will also allow you the opportunity to fabricate the prototype for your aircraft and develop the experience of transitioning ideas to "hardware" and then validate the hardware with prototype flight testing.

PROBLEM STATEMENT

The project goal will be to design a commercial transport which will provide the greatest potential return on investment in a new airplane market. Maximizing the profit that

your airplane design will make for your customer, the airline, will be the design goal. You may choose to design the plane for any market in the fictitious world from which you believe the airline will be able to realize the most profit. This will be done by careful consideration and balancing of the variables such as the number of passengers carried, range/payload, fuel efficiency, production costs and maintenance and operation costs.

REQUIREMENTS

1. **Develop a proposal** for an aircraft and any appropriate derivative aircraft which will maximize the return on investment gained by the airline through careful consideration and balance of the number of passengers carried, the distance traveled, the fuel burned and the production cost of each airplane. The greatest measure of merit will be associated with obtaining the highest possible return on investment for the airline. You will be expected to determine the "ticket costs" for all markets in which you intend to compete. The proposal should not only detail the design of the aircraft but also **must identify the most critical technical and economic factors** associated with the design.

2. **Develop a flying prototype** for the system designed above. The prototype must be capable of demonstrating the flight worthiness of the basic vehicle and flight control system and be capable of verifying the feasibility and profitability of the proposed airplane. The prototype will be required to fly a closed loop figure "8" course within a highly constrained envelope. A basic test program for the prototype must be developed and demonstrated with flight tests.

1.1.2 Mission Definition

Upon examining the Request for Proposal (RFP), it was decided that the easiest way to begin the design of an aircraft would be to define certain goals and objectives in advance. Based upon the RFP and studies of the AeroWorld market, Alpha Design came up with four areas that would be critical to the design of our aircraft. These areas are the primary market, performance objectives, existing restrictions and safety considerations.

1.1.2.1   Primary Market

We have identified the primary market as the one which services the three northern continents in Aeroworld. This decision was based on both the number of passengers

travelling in that region and also the close proximity of these continents to one another. Choosing this market accomplishes two things. First, since the greatest percentage of passengers is located in this area, empty seat space will be minimized. Second, since the distances from city to city are approximately the same (± 1000 feet), the fleet of aircraft required can be standardized, thus reducing operating costs and increasing efficiency. Our market is also influenced by the competition of rail and ship travel. Travel by these two modes has several disadvantages. One is the lengthy travel time, but more importantly, rail travel is limited to land, while ship travel is limited to the seas. Each pose the problem of additional transportation required to reach a final destination. For example, to travel from city J to city N, a train from J to city I is required. Then a ship from that port city to city M is necessary. Finally, a train must be taken from M to the final destination of city N. Air travel eliminates the additional hassle of interconnecting modes of transportation by allowing for direct travel from one city to another.

1.1.2.2   Performance Objectives

Based on our market evaluation, we set some objectives for our desired performance requirements. The range of the aircraft was set such that we could reach any city in our target market with one stop or less. The range is also adequate so the aircraft can reach an alternate city in case of an airport closing due to inclement weather or other reasons. A third factor that influenced the range of the aircraft was the location of the cities in our target market. By specifying a maximum range early in the design process, Alpha Design designed a plane to service both the shorter "commuter" type flights and the longer intercontinental flights which comprise a large portion of the traffic in AeroWorld.

Another factor which needed to be considered was the passenger capacity of the aircraft. This was done by closely examining the passenger loads from city to city in AeroWorld. The capacity must be set such that the aircraft is not too large nor too small. A larger capacity aircraft would result in excessive empty seats which translates into a loss of revenue. A smaller capacity aircraft would result in increased operating costs and inability to handle any growth in the market without the need for construction of derivative aircraft or new designs.

1.1.2.3   Existing Restrictions

Another influencing factor on our design was existence of restrictions. These ranged from gate size availability to speed. All airports have gates that can accommodate a

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five foot wingspan and most airports have gates that accommodate both a five and a seven foot wingspan. Another restriction we faced was limited runway length. Most airports have a runway length of 75 feet but we are servicing one airport with a runway of 60 foot length.

Certain performance requirements have been set as well. The RFP states that the design aircraft must have a level turning radius of 60 feet or less. The RFP also stipulates the speed of sound in AeroWorld to be 35 feet per second. *The Behemoth Apteryx* design satisfies all of these requirements.

1.1.2.4 Safety Considerations

Since this aircraft is a passenger aircraft, safety became an important consideration. Therefore, during the design process all work was done with safety in mind. This involved the use of realistic factors of safety in much of the analysis.

1.2 Design Requirements and Objectives

With the above considerations in mind, we set certain requirements and objectives for our aircraft. Examining the passenger load data given we determined that an optimum passenger load would consist of 50 people. This was decided given that the market was easily divisible by 50 and made flight scheduling much easier. Also, this choice minimized the number of empty seats for flights into those areas where the passenger load was less than 50.

Examining the possible route structure for our target market in AeroWorld, we decided a range of 8000 feet would be adequate for our aircraft. This distance includes redirect and loiter time if necessary.

In order for our aircraft to be successful, certain driving factors were placed to pace the design. Therefore, we decided that the aircraft should be as simple as possible to keep the construction and maintenance costs down. To achieve this goal, fabrication of the aircraft needed to be as simple as possible. This meant that we would use a rectangular wing (all ribs the same), flat plate empennage (simple truss construction, no camber) and a box fuselage (simple truss structure). In the same vein, the airfoil section used for the wing required as flat a lower surface as possible. This would help to reduce tooling costs.

As noted above, there are two gate sizes in AeroWorld airports, five foot and seven foot. To be able to access all the gates, our design will employ a five foot span. If a larger span was desired, the aircraft would only be able to use the seven foot gates unless the

wings were hinged in some way. This adds unnecessary complexity to the design and goes against one of our major design objectives.

Since one of the runways in AeroWorld is 60 feet in length, it was desirable for our aircraft to have a take-off roll of 50 feet or less. This was for safety considerations. As a result of this requirement, the powerplant must be capable of supplying sufficient thrust to take off. In addition to this requirement, the weight needed to be kept as low as possible so that the necessary lift may be generated.

The final design of our aircraft met all of these objectives. The final flight tests will determine our success.

2.0 CONCEPT SELECTION STUDIES

As individual members of Alpha Design interpreted the RFP, several concepts were considered before the final concept was selected. Each of these individual concepts had some points which were desirable and the best features of all the concepts were incorporated into the final concept.

2.1 Concept #1

This concept was one of the more radical concepts of the design. (See Figure 2.1) This aircraft utilized a high mount wing with a span of 60 inches to meet gate requirements in Aeroworld. The leading edge has a sweep angle of 8°. This will allow a more elliptical configuration to reduce induced drag. The root chord is 12 inches while the tip chord is 8 inches leading to a taper ratio of 0.667. The aspect ratio is 6.

The fuselage contains the passenger cabin, cockpit and avionics suite, as well as the fuel source (batteries). The passenger capacity is 50. The seating configuration has two seats on the left side (front view), an aisleway, and a single seat on the right, except for the first row, where an entry is located.

There are two vertical tails in this design. This is to increase the vertical tail area without having an inordinately large tail section. There is a single horizontal tail connecting the two stabilizers.

The aircraft also utilizes two engines to provide more power for take-off and to account for the larger size.

2.2 Concept #2

This concept was much more conventional in nature and played a large role in the selection of the final concept. (See Figure 2.2) The wing is mounted on top of the fuselage and has a span of seven feet, a root chord of 12 inches and a tip chord of 8 inches. The fuselage is structured such that the passenger capacity will be 50 and there will be sufficient room for luggage and galley space. As in the first concept, the fuel and avionics package are contained in the fuselage.

The aircraft utilizes a single engine mounted in the nose of the aircraft in a traditional "puller" configuration. This also helps to streamline the aircraft.

The main control surfaces for this concept will be the elevator and rudder. Since ailerons are not present, the aircraft will use sideslip for a coordinated turn. As a result of this there will be approximately 7° of dihedral in the wing for roll stability.

2.3 Final Concept

Based upon the above concept studies, a final concept could be prepared for our design. Drawing on the DR&O, the wing was decided to have a span of 60 inches and a chord of 14 inches. The wing will be top mounted with 2° dihedral in the wing. The fuselage is 44 inches long, 7 inches wide, 5 inches high at the highest point and 2 inches high at the tail.

The aircraft will utilize a single engine mounted in the nose of the aircraft for the same reasons as mentioned above. Cooling of the engine will be done by airflow across the engine and avionics. This is accomplished by placing vents in the front and rear of the aircraft. Control of the aircraft will be obtained through the use of ailerons, elevators, and a rudder. For the final design, a conventional "tail- dragger" set up for the landing gear will be employed.

The fuselage will be constructed such that the passenger capacity will be 50. Additionally, the fuselage will provide ample space for luggage, passenger facilities, and the aircrafts' mechanical components.

Based upon further study, the airfoil section used will provide as high a maximum lift coefficient as possible while still keeping a flat lower surface.
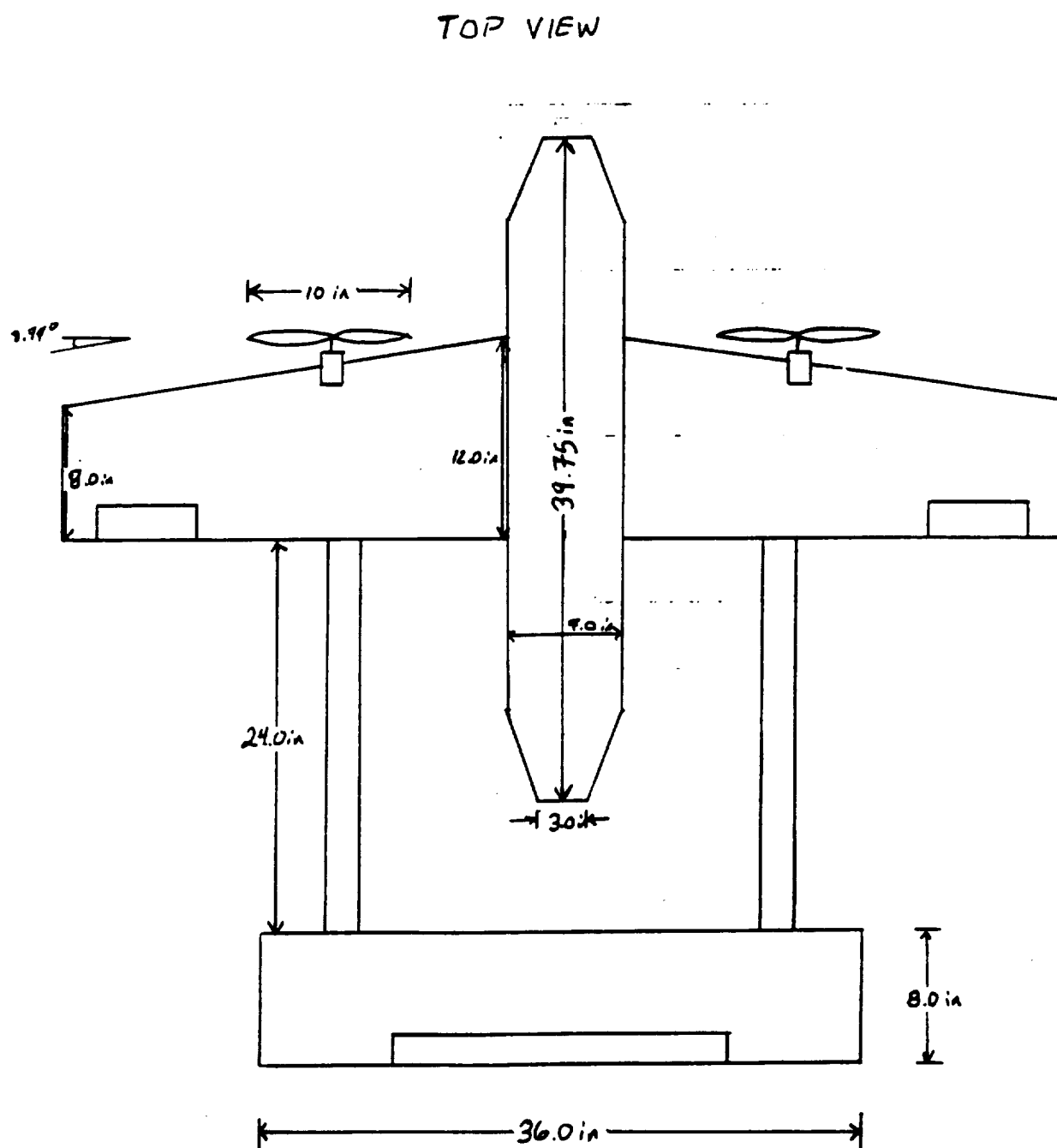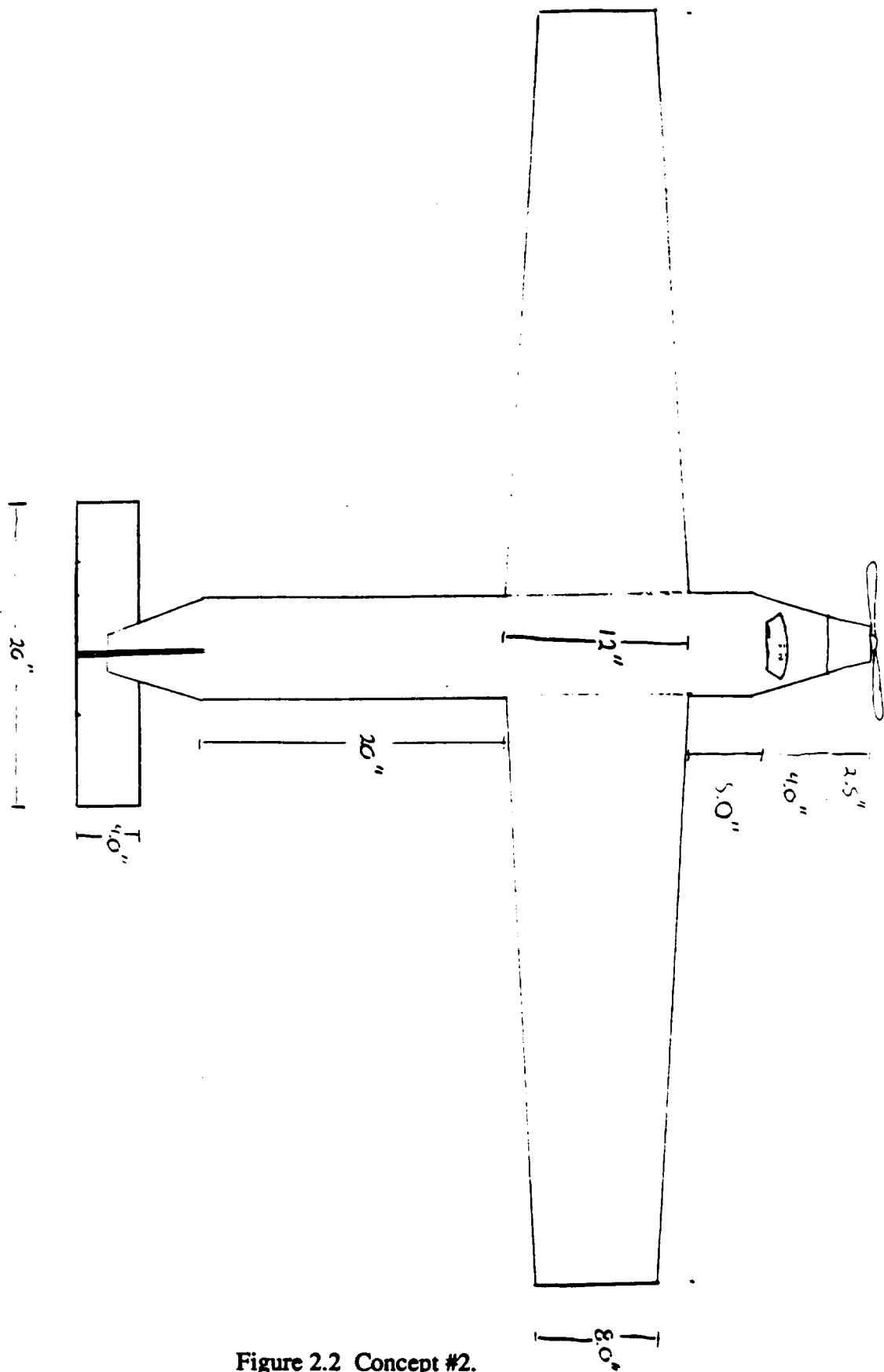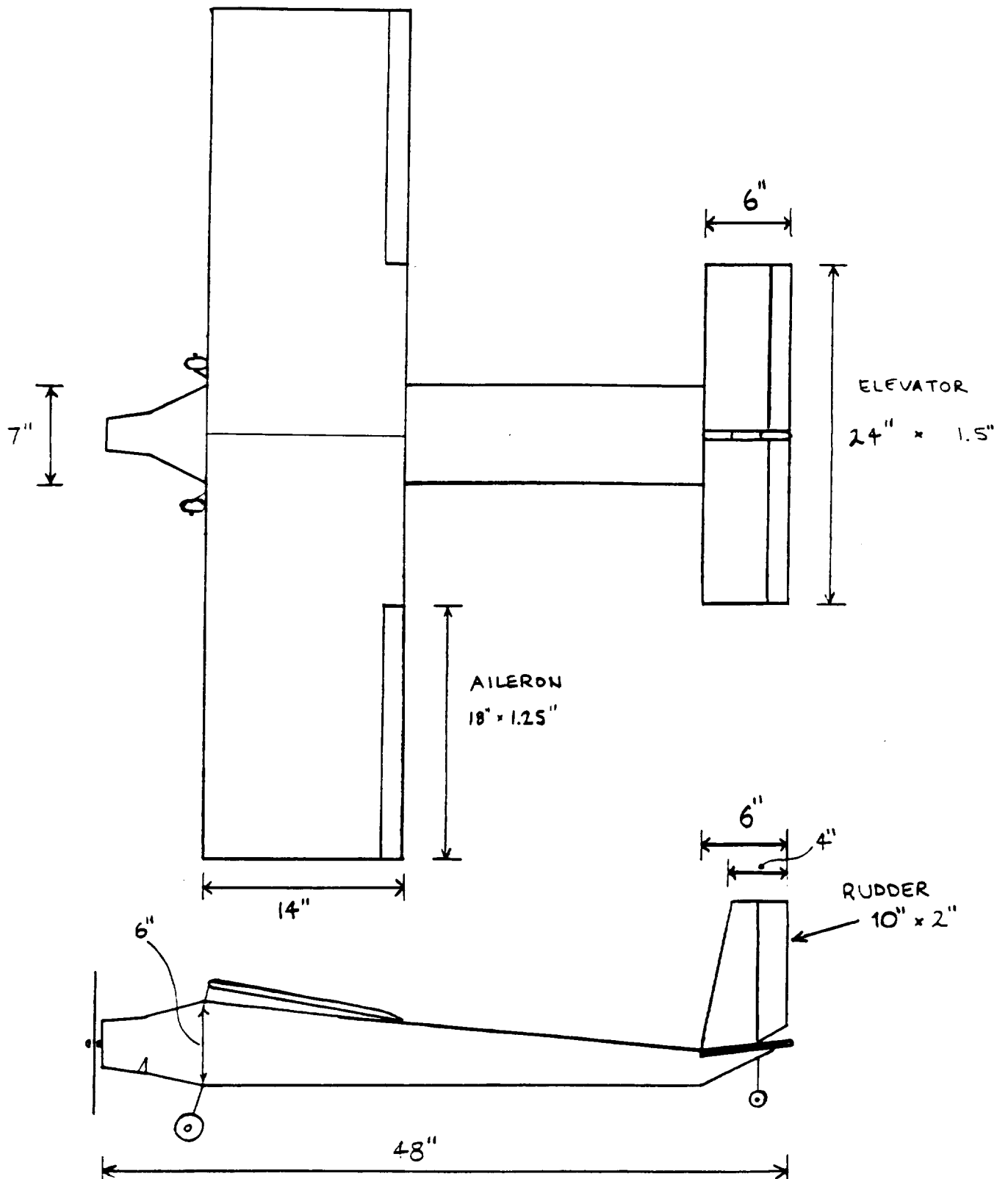
TOP VIEW

Figure 2.1 Concept #1.

Figure 2.2 Concept #2.

16"

60"

2"

6"

ELEVATOR

24" × 1.5"

7"

AILERON

18" × 1.25"

14"

6"

4"

RUDDER

10" × 2"

6"

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48"

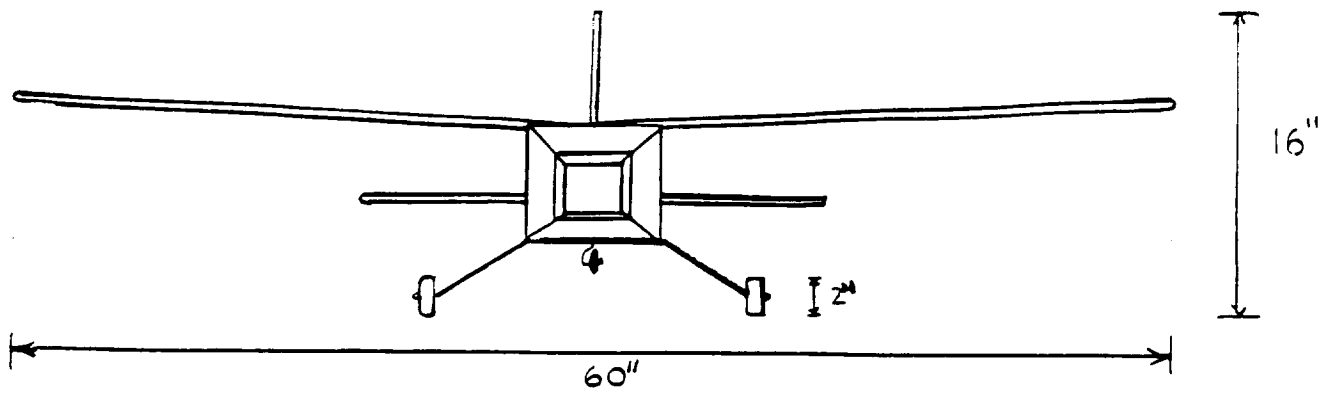## 3.0   AERODYNAMIC DESIGN

3.1   Airfoil Selection

In the selection of the airfoil for our wing, our first criterion was that it have a high lift-to-drag ratio. The fundamental purpose of an airfoil is to produce lift with as little drag penalty as possible, the lift-to-drag ratio measures how well the airfoil accomplishes this. Secondly, the airfoil had to have a flat bottom and have no concave curves in order to reduce construction time and cost as per our self imposed requirements. The airfoil had to perform well in the low Reynolds number range, since the speed is limited to 35 ft/s. Finally, since our limited span leads to a small wing and relatively high wing loading, the airfoil must be able to deliver a high maximum C_l so that the wing can generate enough lift for takeoff.

Low Reynolds number airfoil test data were analyzed, and the airfoils which best satisfied our L/D and C_{lmax} criteria were: Eppler 387, S2091, S3021, SPICA, and Wortmann FX137. These airfoils all performed well at Re equal to 200,000, which corresponds to our flight range, but their drag increases substantially at lower Reynolds numbers due to the formation of separation bubbles.

Overall, the SPICA airfoil met all of our criteria very well. It has a high maximum lift-to-drag ratio of 67. Furthermore, this ratio remains high over the range of lift coefficients at which the airfoil will be operating during flight. SPICA's maximum C_l value of 1.4 is higher than most of the other airfoils we examined. The airfoil's performance is poor at values of C_l below .4 (see Figure 3.1), but with a cruise C_l of .6 computer analysis has shown that the section lift coefficient will be above .4 over most of the wing during flight (see Figure 3.2).

3.2   Wing Design

The wing of our airplane was designed to be simple, once again to facilitate construction; it has a rectangular planform and no twist. It will have a 2° dihedral to give the plane static roll stability. With the span limited to 5 feet, a chord of approximately 1 foot is necessary to make the wing loading reasonable. Since a higher aspect ratio lowers the induced drag on the wing, we desired to keep the chord as short as possible. This was limited because, as mentioned above, the airfoil's drag is very high below a Reynolds number of 200,000. Figure 3.3 shows that a chord of 14 inches allows us to fly at speeds as low as 30 ft./s. while keeping Re above 200,000.

Increasing the chord decreases the slope of the lift curve (see Figure 3.4); this means that the wing must be at a larger angle of attack to generate the same lift. In order to cruise with the airplane level, the wing is mounted at a 7 degree angle of attack. The aspect ratio of the wing is 4.3, and the wing loading is 10 oz./ft^2. The maximum lift-to-drag ratio is 8, occurring with the plane oriented at 4 degrees. The lift curve for the entire aircraft can be seen in Figure 3.5.

Since the aspect ratio is relatively low, we considered the addition of winglets to the wing. We had no expertise in designing winglets, but we used the LinAir computer program to examine the drag reduction realized by using different winglet configurations modeled after those on existing airplanes. However, the change in the aircraft drag polar was not noticeable, so this idea was not pursued further.

3.3   Drag Prediction

In order to estimate the drag characteristics of the aircraft, it is assumed that the drag can be expressed in the following form:

$$C_D = C_{D_0} + \frac{C_L2}{\pi \, AR \, e}$$

Two methods were used to estimate the zero lift drag coefficient, C_{D_0}: the Equivalent Skin Friction method from Jensen's thesis, and Nelson's Subsonic Drag breakdown method.

Jensen's method provides an estimate of C_{D_0} by assuming that the aircraft parasite drag during cruise is predominantly due to skin friction. The aircraft is broken into component parts and the following equation is used:

$$C_{D_0} = \frac{C_f \, S_{wetAC}}{S_{ref}}$$

where S_{wetAC} is the total wetted area of the aircraft. Jensen recommends using C_f of .0055 for this type of aircraft, and Table 3.1 shows the wetted areas for each component. This method yields an estimated C_{D_0} of .02.

Component	S_{wet} (in^2)
Fuselage	1110
Wing	840
Horizontal Tail	160
Vertical Tail	50
Landing Gear	840

Total Aircraft	3060

Table 3.1. Component Wetted Areas.

The second method is a bit more sophisticated and does not depend solely on skin friction. This is Nelson's Subsonic Drag Breakdown method. Once again, the aircraft is broken down into separate parts, and then the C_{D_0} for each component is estimated. For the fuselage, this was found using a method in Fluid Dynamic Drag based on the frontal area of the fuselage and the skin friction over the body:

$$C_{Dofuse} = .44 \left(\frac{1}{d}\right) + 4 C_f \left(\frac{1}{d}\right) + 4C_f \left(\frac{1}{d}\right)^{1/2}$$

where l and d are the length and maximum diameter of the fuselage, respectively. For the wing and tail, the C_{D_0} values were found from the airfoil drag polars (see Table 3.2). Nelson's handout recommended using a value of .017 for the landing gear. These component C_{D_0} values are multiplied by their individual reference areas, summed and divided by the aircraft reference area:

$$C_{D_0} = \frac{\Sigma \, S_\pi \, C_{D_\pi}}{S_{ref}}$$

This method yielded a C_{D_0} estimate of .06.

Component	S_π (in^2)	C_{D_π}	$S_\pi C_{D_\pi}$
Fuselage	42.0	0.218	9.156
Wing	840.0	0.032	26.88
Horizontal Tail	160.0	0.0012	0.1792
Vertical Tail	50.0	0.0012	0.056
Landing Gear	840.0	0.0170	14.28
		$\Sigma \, S_\pi \, C_{D_\pi} =$	50.5512

Table 3.2. Component Breakdown.

In our calculations we used the average of these two different C_{D_0} estimates, .04. The skin friction method is too optimistic; it does not take into account the pressure drag, which would be important for a plane with such a wide fuselage. Conversely, the drag breakdown method's estimate is probably pessimistic because the value we obtained for the fuselage C_{D_0} was much higher than the typical value for a prop driven plane given in Nelson' paper.

Finally, the drag polar was completed by estimating the Oswald efficiency factor, e. Another breakdown method was used:

$$\frac{1}{e} = \frac{1}{e_{wing}} + \frac{1}{e_{fuse}} + \frac{1}{e_{other}}$$

Design charts from Nelson were used to estimate the efficiency factors for the components. This yielded an e for the airplane of .76. The airplane drag polar can be found in Figure 3.6.
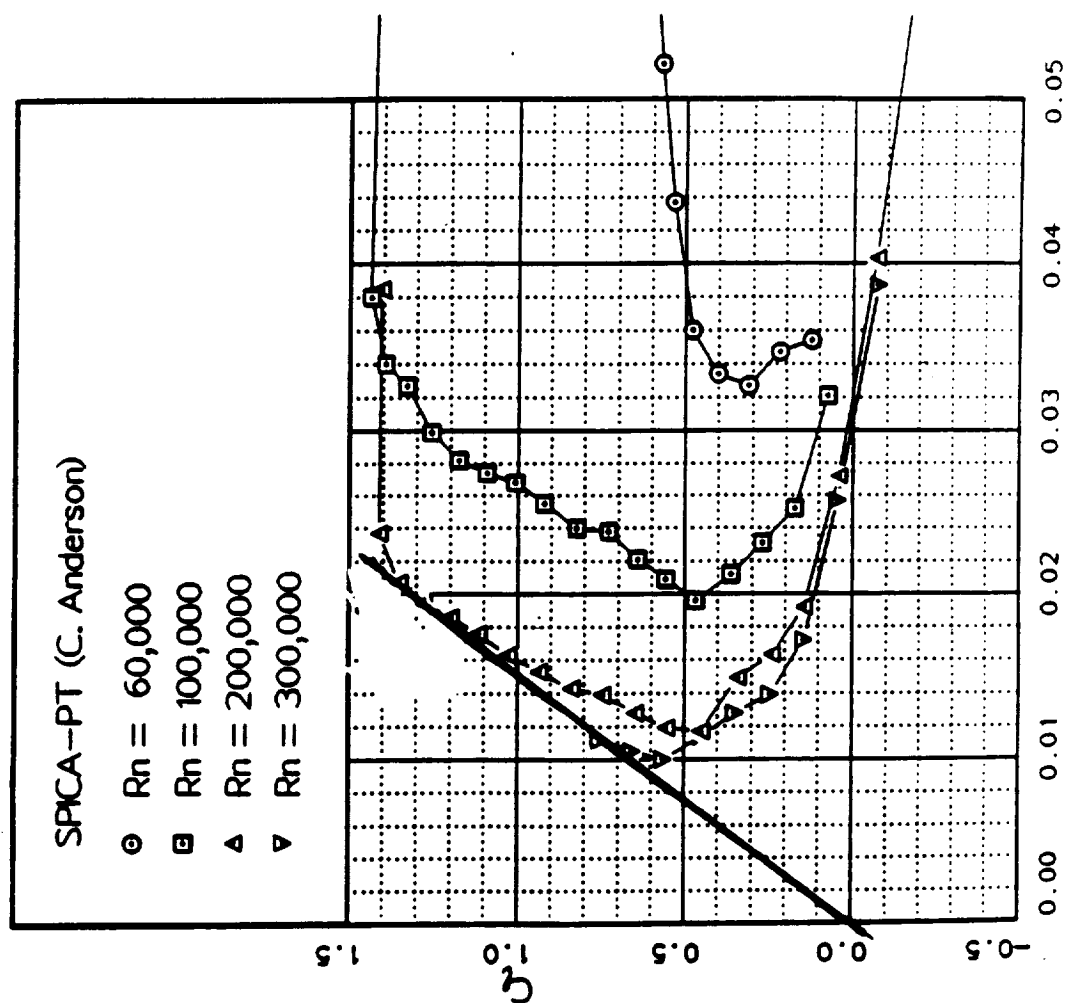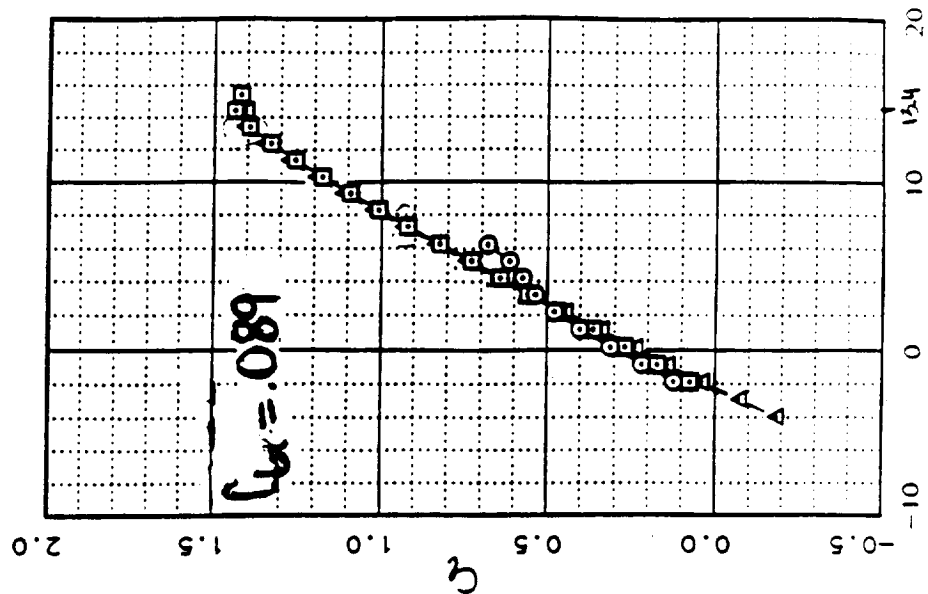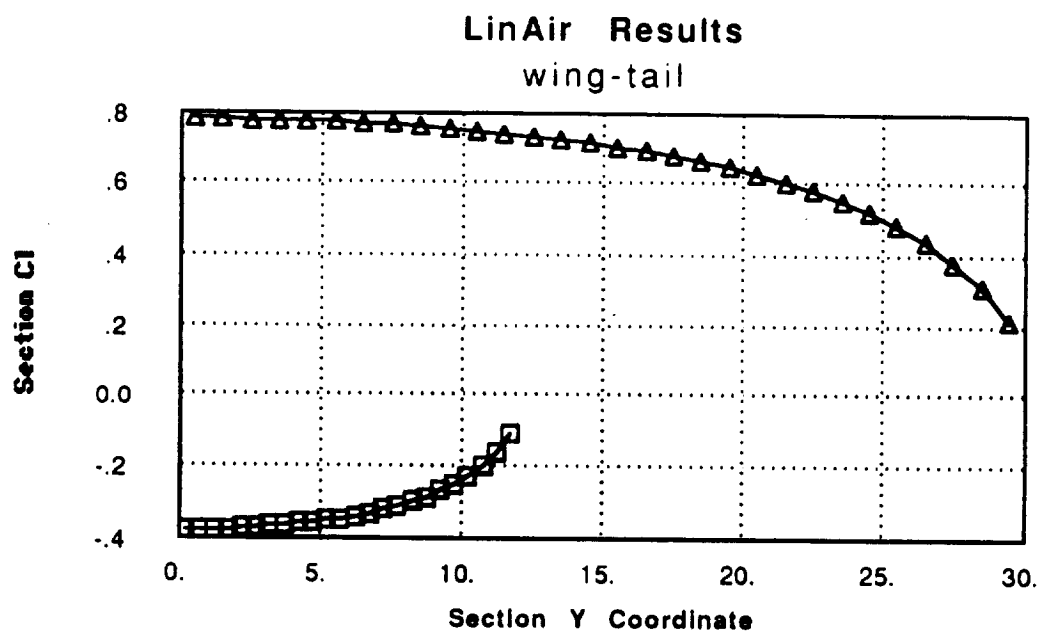
Figure 3.1. SPICA Airfoil Characteristics.
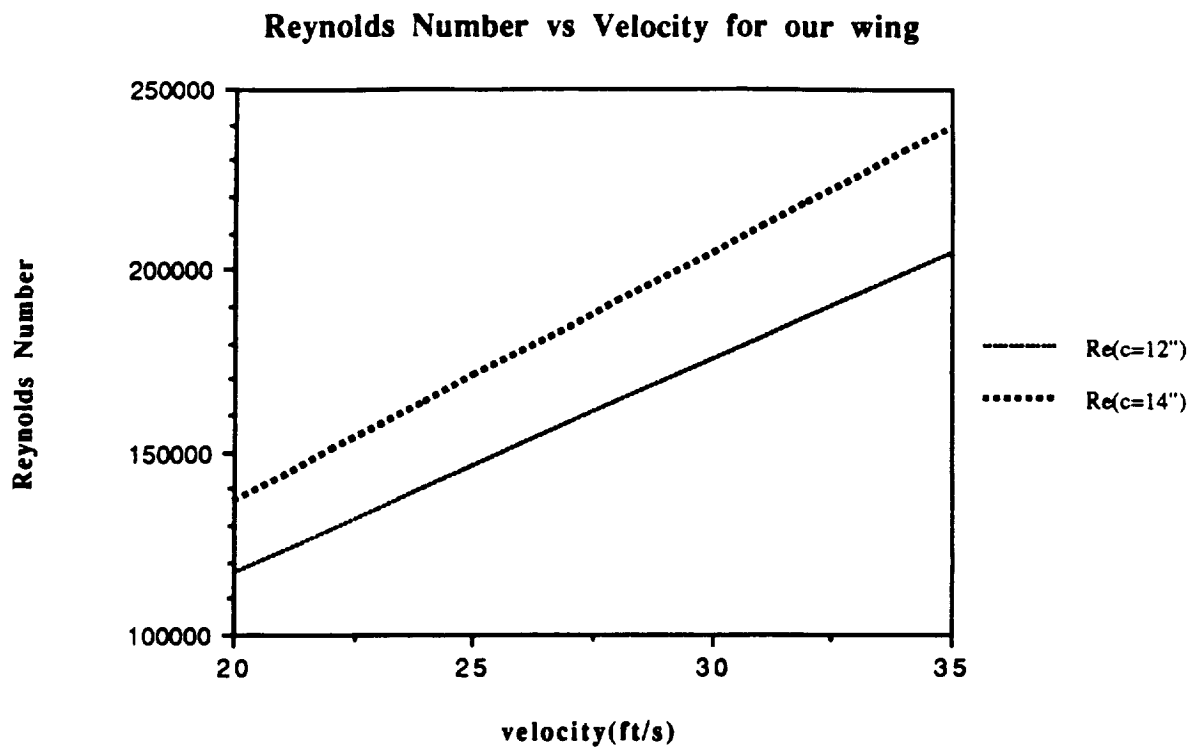
Figure 3.2. Lin-Air Computer Generated Lift Distribution.

Reynolds Number vs Velocity for our wing

Figure 3.3. Effect of Chord Length on Reynolds Number.
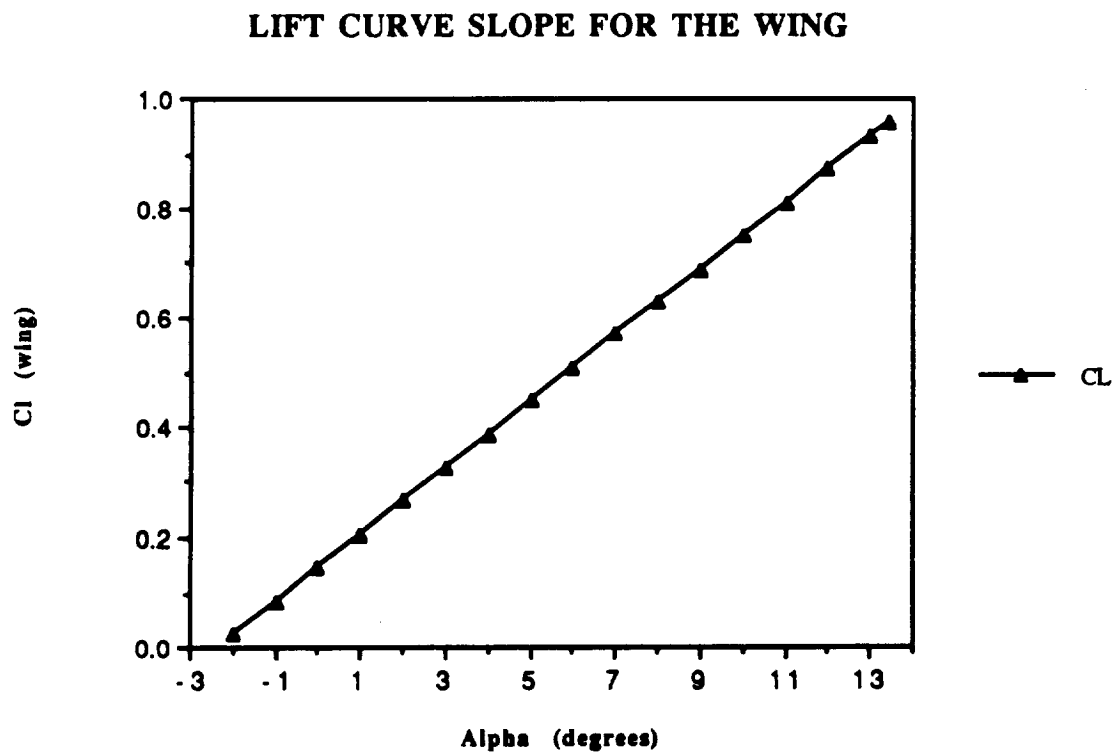
LIFT CURVE SLOPE FOR THE WING

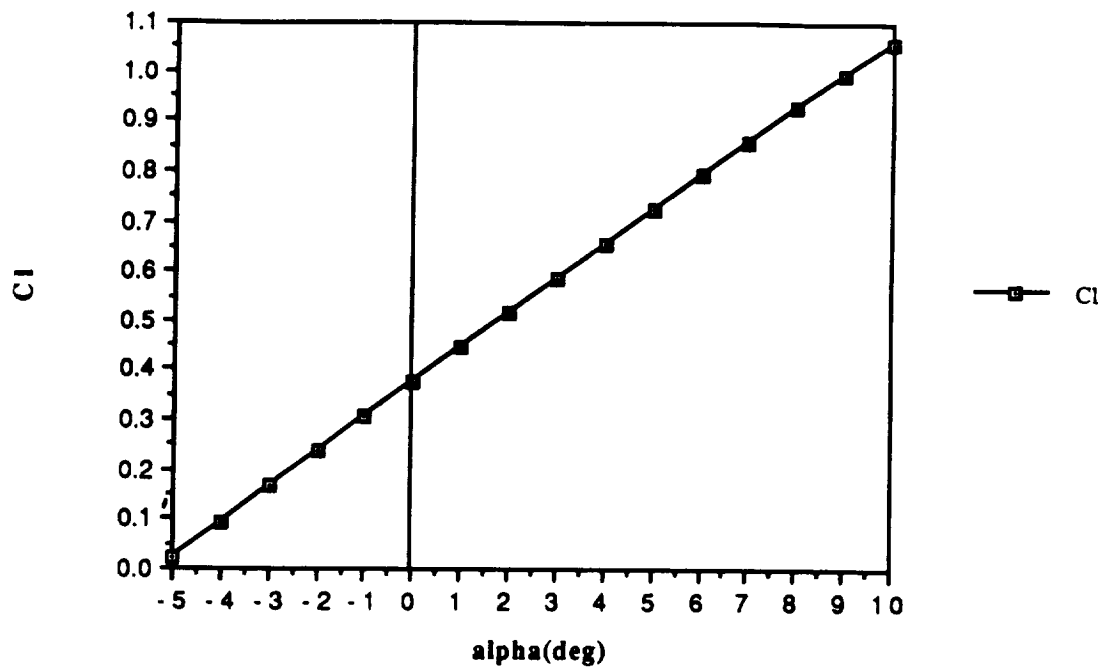Figure 3.4. Wing Lift Curve.

Lift Slope for Aircraft

Figure 3.5. Aircraft Lift Curve.
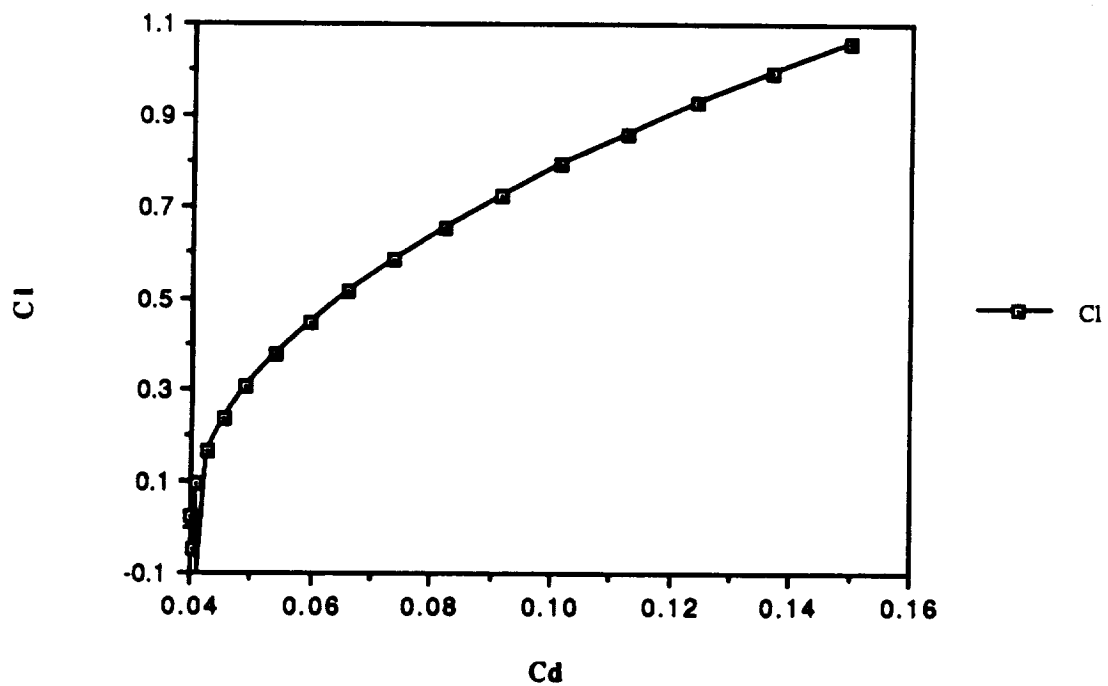
Drag Polar for Aircraft

Figure 3.6. Drag Polar for Aircraft.

4.0 STABILITY AND CONTROL

After determining the final concept for the *Behemoth Apteryx* and choosing the airfoil type and dimensions, it was necessary to design a control system to insure stability and control of the aircraft during flight. To design this system, all control modes had to be considered--pitch, roll and yaw. The design process was to determine the minimum surface sizings to maintain stability, the proper mounting angles if any, and then the necessary increases in sizes to achieve our desired stability characteristics. Once the necessary stability surfaces were sized, then the control surfaces were determined in order to give adequate maneuverability to our aircraft.

4.1   Horizontal Tail

The first task was to size the horizontal tail for longitudinal static stability. A flat plate airfoil was chosen for ease in construction in line with our design objectives. The horizontal stabilizer was then sized by calculating the moments about the c.g. caused by the wing and fuselage and then sizing the stabilizer to counter these moments. The plot of the C_{m_α} curve for the plane must be negative for static stability, so the size of the stabilizer was chosen to give a zero slope of the C_{m_α} curve which would be the absolute minimum size necessary. It was then oversized to give the desired magnitude of the C_{m_α} curve slope and to match a V_H of between .4 and .5 as suggested by Mr. Joe Mergen. He said that at least that much would be needed to maintain stability.

The governing equation for the C_{m_α} curve is

$$C_{m(plane)} = C_{m_0} + C_{m_\alpha}\alpha$$

and since C_{m_0} must be positive (in order to trim the aircraft at positive angles of attack), the C_{m_α} must be negative. C_{m_α} for the plane was determined by summing the contributions of each major component of the plane. Since the C_{m_α} for the wing was .004832/° (positive) as given from the aerodynamics group and for the fuselage it was .0000402/° (also positive) as calculated in Table 4.1, the $C_{m_{\alpha t}}$ needed to be negative to give the zero slope for minimum area. Using these two values, the minimum surface area of the horizontal tail was calculated at 49 in^2.

This minimum area was very small in comparison with experience and common sense. As stated before, the size needed to be increased so that the C_{m_α} curve slope would be negative. We tripled the minimum size, which gave us a V_H in our desired

range, and then made the tail 24.5 in. x 6.5 in. for simplicity,. These dimensions also gave an aspect ratio of 3.76--within the range of 3 to 5 suggested in Entering Electrics. The values of C_{m_α} at the various c.g. locations are listed in Table 4.2. We found that C_{m_α} was most strongly a function of c.g. placement, especially when finding the minimum control surface area to maintain stability. Moving the c.g. just 2.5 inches back from the leading edge of the wing increased the minimum size necessary by 2 times.

Station	Δx	W_f	x_i	$\bar{x}$	$\dfrac{x_i}{Cr_e}$	$\dfrac{x_i}{l_h}$	$\dfrac{\partial \varepsilon_u}{\partial \alpha}$	$W_f{}^2\dfrac{\partial \varepsilon_u}{\partial \alpha}\Delta x$
1	1	2.42	5.5		.393		1.50	8.785
2	1	3.25	4.5		.321		1.57	16.583
3	1	4.09	3.5		.250		1.65	27.601
4	1	4.92	2.5		.179		1.85	44.78
5	1	5.75	1.5		.107		2.25	74.39
6	1	6.59		1.0	.071		4.0	173.71
7	3	7				.0549	.054	7.94
8	3	7				.165	.164	24.108
9	3	7				.274	.272	39.98
10	3	7				.384	.381	56.01
11	3	7				.494	.490	72.03
12	3	7				.604	.599	88.05
13	3	7				.713	.707	103.93
14	3	7				.823	.816	119.95
15	1	7				.896	.888	130.54

$$\sum W_f{}^2\frac{\partial \varepsilon_u}{\partial \alpha}\Delta x = 988.39 \text{ in}^3 \qquad \therefore \ C_{m_{\alpha f}} = .002303/\text{rad} = .0000402/^\circ$$

Table 4.1 Calculation of $C_{m_{\alpha f}}$

c.g placement	C_{m_α}
.28c(most forward)	-.0228/$^\circ$
.33c(optimal)	-.0198/$^\circ$
.45c(most aft)	-.0125/$^\circ$

Table 4.2 C_{m_α} for c.g. placements

As stated before, the C_{m_0} for the entire airplane must be positive in order to trim the aircraft at positive angles of attack. The C_{m_0} was found by summing the contributions of each airplane component. The C_{m_0} of the wing was obtained from the aerodynamics group and the C_{m_0} of the fuselage was calculated by the contributions of

each station on the fuselage as shown in Table 4.3. In calculating this, it was found that the tail had to be mounted at a negative angle of attack in order to satisfy the equation

$$Cm_{O_t} = V_H * C_{L_{\alpha_t}} * (\varepsilon_0 + i_w - i_t)$$

and still have a positive Cm_O. In order to trim in our desired range, we decided to mount our tail at -3°. The variation of C_m with different i_t's and at the forward, aft, and optimal c.g. positions is shown in Figures 4.1-4.3.

Our optimum c.g. placement is at .33c. This gives us a static margin of 19% which is slightly high, but acceptable. This is with our neutral point at .52c. It was further verified that our aircraft would be stable with a c.g. travel from .28c to .45c. The reason that our aft most c.g. placement is only at the .45c while the neutral point is at .52c is to insure stability. The slope of the C_{m_α} curve approaches zero and this severely compromises our stability.

Station	Δx	W_f	i_f	$W_f^2[\alpha_{0w}+i_f]\Delta x$
1	1	2.25	0	−12.15
2	1	2.75	0	−18.15
3	1	3.25	0	−25.35
4	1	3.75	0	−33.75
5	1	4.25	0	−43.35
6	1	4.75	0	−54.15
7	14	5	0	−840
8	4	4.8	2.86	41.93
9	4	4.4	2.86	35.24
10	4	4.0	2.86	29.12
11	4	3.6	2.86	23.59
12	4	3.2	2.86	18.64
13	5	3.0	0	−108

$$\sum W_f^2[\alpha_{0w}+i_f]\Delta x = -986.39 \qquad \therefore \quad Cm_{O_f} = -.002183$$

Table 4.3 Calculation of Cm_{O_f}

4.1.1 Stability Curves

Our final decisions on i_t, c.g. placements, and horizontal tail size allowed us to calculate the various C_{m_α} curves for each c.g. position. Figure 4.7 is a plot of the C_{m_α} curves for the most forward, most aft, and optimal c.g. positions.

4.2 Vertical Tail

With the horizontal tail sized and longitudinal stability verified, the next task was directional stability. The analysis to determine the minimum size for the vertical tail for stability showed that it, too, was extremely small. This was to be expected since for the vertical tail there was no wing moment to counteract, only the fuselage and possibly the torque of the motor. Thus the analysis took another direction. The rudder will be used for coordinated turns (i.e. with ailerons) as well as directional stability. Our final design was determined by closely matching the sizing proportions of successful past aircraft and by trying to achieve a V_V of .2 as suggested by Mr. Mergen. This gave us a tail with a root of 6 in. and a tip of 4 in. with a height of 10 in. This gives our aircraft a value of C_{N_β} of -.0000185/°. This is an oversized vertical tail, but the C_{N_β} shows that we will have adequate directional stability.

4.3 Control

With the stabilizers sized, the control surfaces were determined. The first control surface considered was the elevator. The elevator was determined by selecting various area ratios and then calculating the effectiveness for various elevator deflections. The final design chosen was one with a S_e/S_t of .25 which was an elevator of chord length 1.25 in. running the length of the horizontal tail span. This gave us a $C_{m_{\delta e}}$ of -.923 which gives us a trimmed aircraft at approximately 0° elevator deflection. The elevator effectiveness curves for the c.g. placements are given in Figures 4.4-4.6.

The rudder was again sized by comparison with successful aircraft. It is designed to be 1.75 in and run the height of the vertical tail. The rudder effectiveness is .0225/°. This compared well with previous values and was not changed.

4.4 Ailerons

Roll control and stability both are achieved by means of ailerons. The ailerons were sized at 18 in. by 1.5 in. which gives a $C_{l_{\delta a}}$ of .0011. This was used to calculate the change in lift needed to roll the plane to the necessary 37.5° bank angle for a 60 ft. turn radius with an aileron deflection of 10°. This was calculated using the moment of inertia about the center line of the fuselage, and a selected roll rate of 1 radian per second. The actual size necessary was 15.87 in. by 1.36 in. but they were slightly oversized to give a tighter turn if necessary. Our plane will roll to 37.5° in .65 seconds for the 60 ft. turn radius and to 42.7° in .75 seconds for a 50 ft. turn radius. Since the ailerons were adequate to roll the airplane for a turn, they were judged to be adequate to provide the necessary compensation for roll stability if needed. Additional roll stability will be provided by 2° of wing dihedral, which was suggested by Joe Mergen.

The actuators will consist of a series of push rods and servo motors, and the servo motors placement will be governed by c.g. location requirements. The control system will need 3 servos and a fourth will be designated for the throttle.
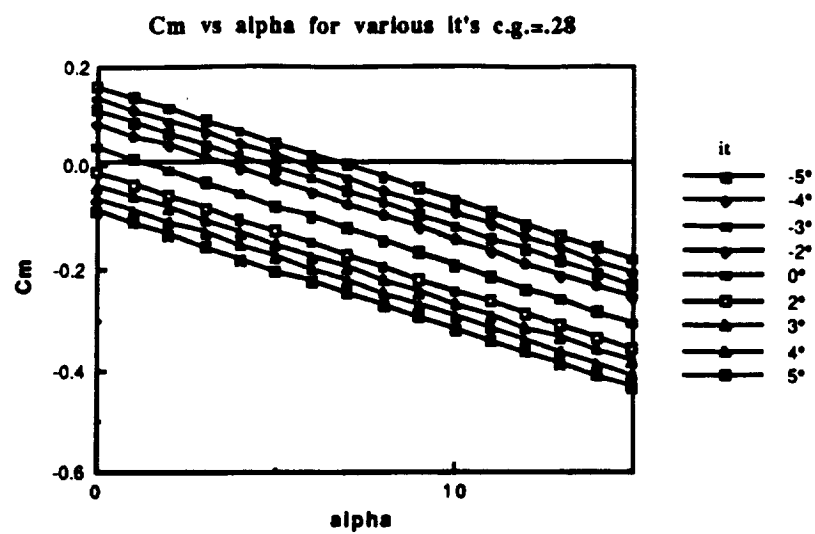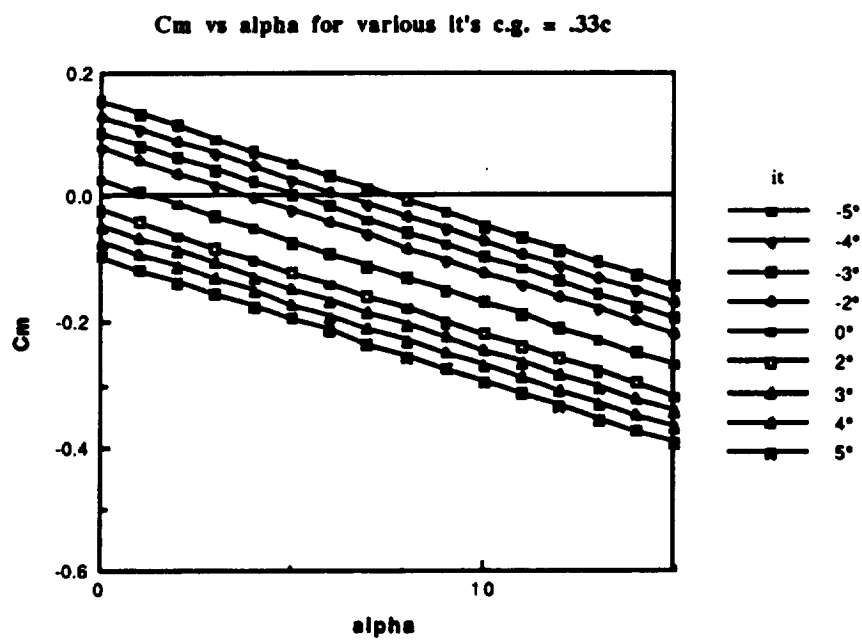
Cm vs alpha for various it's c.g.=.28

Figure 4.1 C_{m_α} for various i_t's at c.g.= .28c

Cm vs alpha for various it's c.g. = .33c

Figure 4.2 C_{m_α} for various i_t's at c.g.=.33c

Cm vs. alpha varying it's c.g.=.45

Figure 4.3 C_{m_α} for various i_t's at c.g. = .45c

Cm vs. alpha for various de's c.g.=.28
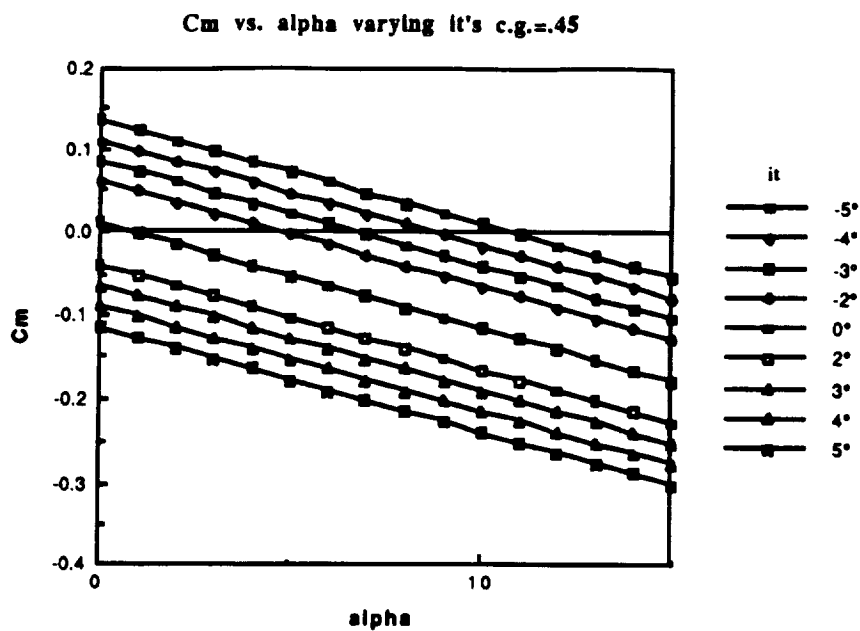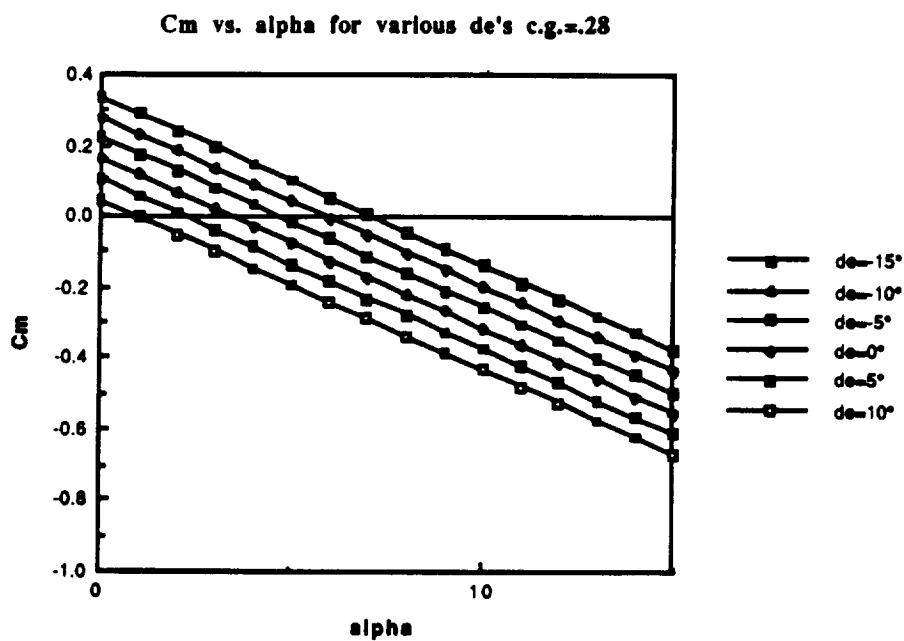
Figure 4.4 Elevator Effectiveness at c.g.=.28c

Cm vs. alpha for various de's c.g.=.33
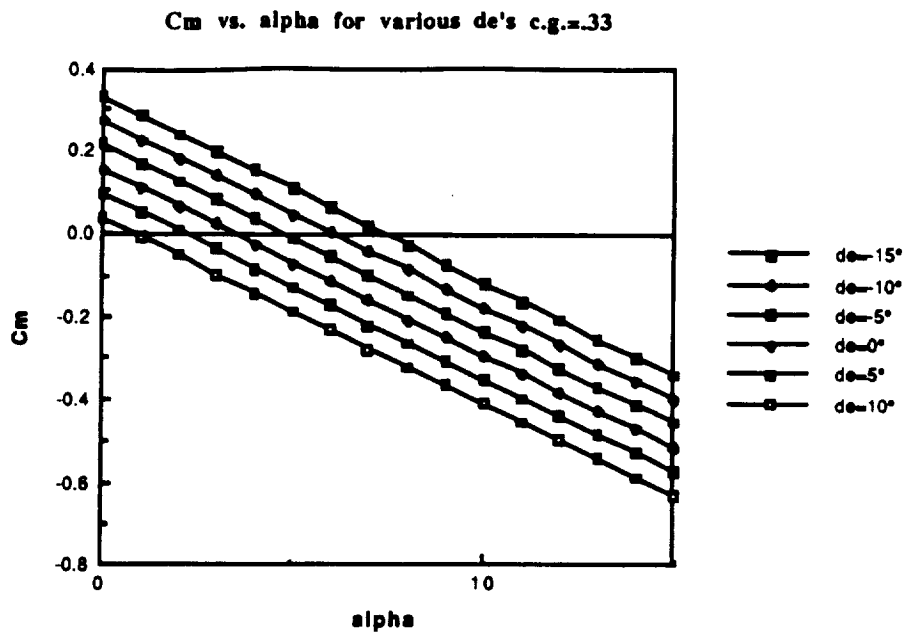
Figure 4.5 Elevator Effectiveness at c.g.=.33c

Cm vs. alpha for various de's c.g.=.45
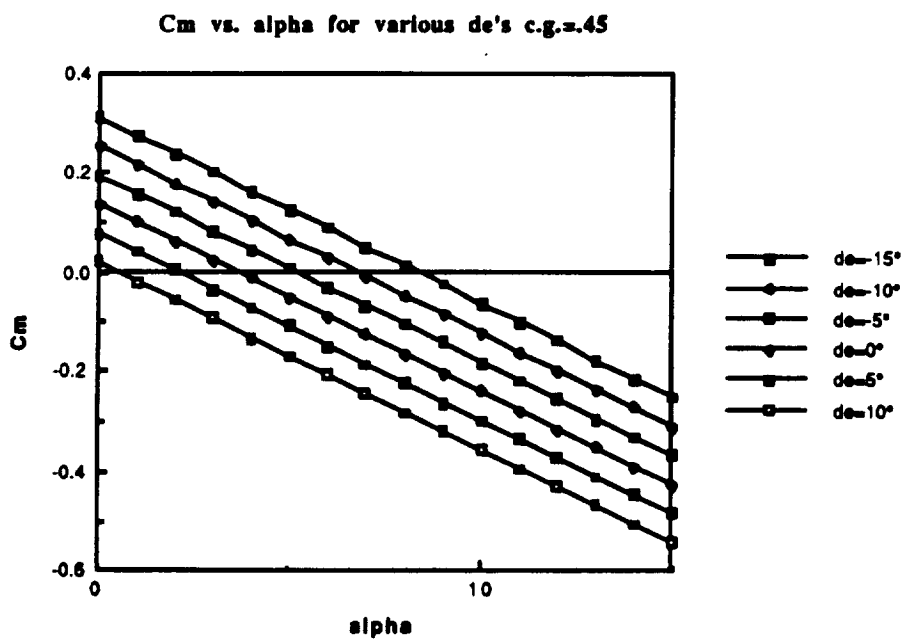
Figure 4.6 Elevator Effectiveness at c.g.=.45c
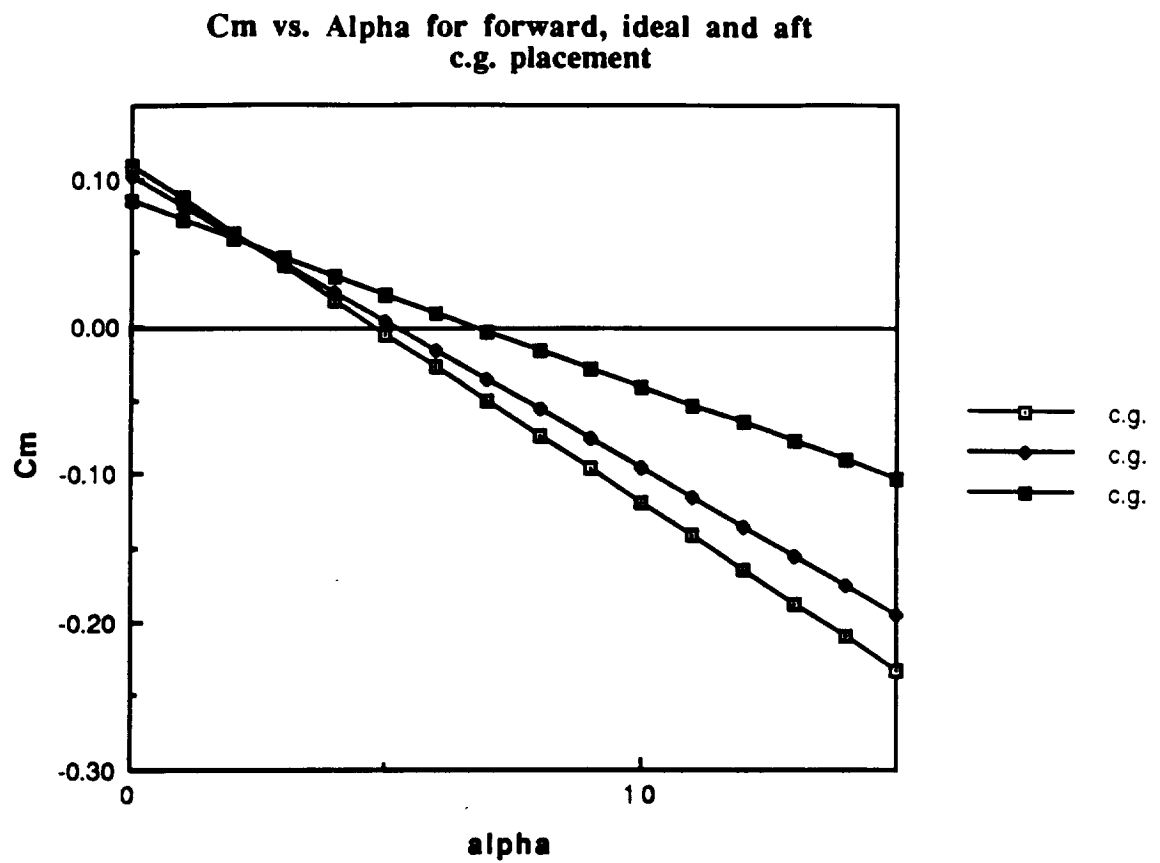
Figure 4.7 C_{m_α} for Most Forward, Most Aft, and Optimal c.g. position

5.0 PROPULSION SYSTEM

As stated in our DR&O, the propulsion system must be capable of providing the necessary thrust for takeoff, given our low planform area and high take-off and stall velocity. Therefore, the choice of a motor and propeller became an even more critical factor in the design process.

5.1 Motor Selection

In keeping consistent with our design objectives, the two factors deemed most important in selecting a motor are minimization of fuel burned, which is primarily a function of propeller efficiency, and low weight. The weight of the motor is especially important because of our small wingspan and high wing loading.

The actual selection of the motor was simplified by examining the aircraft database to see previous designs. This quickly narrowed our choice to the Cobalt Astro-05 and the Cobalt Astro-15. Since the need for adequate power was paramount, cost was considered a secondary factor in choosing the motor. The Astro-15 gives a 100% increase in power over the Astro-05 with only a 1.5 ounce increase in weight. Since the Astro-15 is a larger motor and has a larger current draw, it does require a larger battery pack. However, Alpha Design felt the additional weight justified given the large increase in power. The critical parameters of the propulsion system are summarized in Table 5.1.

Motor	Cobalt Astro-15
Motor weight (including mount)	10.24 oz.
Propeller	Top-Flight 10-4
Prop efficiency at cruise	0.74 (from Apple IIe program)
Prop rpm at cruise	5307
Estimated static thrust	2.6 lb (from database)
Cruise power setting	84%
Cruise range (steady, level flight)	15 455 ft
Battery capacity	650 mah
Battery pack voltage	14.4 (nominal) 16.2 (peak)
Battery pack weight	12.67 oz.

Table 5.1 Propulsion System Parameters.

5.2 Propeller Selection

Fuel efficiency is the main parameter that drives the propeller selection. Since the cruise phase of the flight occupies the majority of the total flight time, the propeller should be optimized for this flight regime. A smaller prop, assuming it provides enough thrust for cruise, would yield the best performance for cruise. However, static thrust increases with propeller diameter. As a result of our objectives, we considered a larger diameter prop to provide enough thrust for take-off.

During the propeller analysis, C_t and C_p as a function of J were obtained for seven different props and compared. Each prop was then evaluated based upon the number of amp-hours burned during the cruise phase. The results of this analysis can be seen in Figure 5.1. The best cruise performance is provided by a Top Flight 9-4 propeller. However, it is interesting to note the performance of the other propellers. There is only a difference of a few milliamp-hours in performance over our target 8000 feet range. Therefore, Alpha Design decided to use a Top Flight 10-4 prop to increase the static thrust at take-off without paying a severe penalty in fuel efficiency. The efficiency of the Top Flight 10-4 as a function of advance ratio is shown in Figure 5-2.

5.3 Battery Pack Selection

In keeping with our objectives, the battery pack needs to be able to provide enough current to power the motor efficiently and also have enough energy capacity to meet our target range. Therefore, all the battery types are rapid charge to provide on-site charging and rapid discharge to provide high current draws during take-off. The critical parameters of the batteries considered are shown in Table 5-2.

Model	Company	Capacity [amp-hrs]	Weight of each cell [oz]	Voltage per cell [volt]
Gates 650-SCR	Gates	0.650	0.92	1.2
P-90SCR	Panasonic	0.900	1.23	1.2
P-120SCR	Panasonic	1.200	1.66	1.2
N-600SCR	Sanyo	0.600	1.02	1.2
N-900SCR	Sanyo	0.900	1.38	1.2
N-1200SCR	Sanyo	1.200	1.84	1.2

Table 5.2 Critical Battery Parameters.

A given battery pack must be able to provide a range of 8000 feet for our mission. Since the aircraft must also have a 1 minute (RWT) loiter capability at a cruise velocity of 32 ft.s, the effective aircraft range must be increased by at least 1920 feet. Take-off must also be factored into the pack selection. Therefore, the battery pack must be able to provide an effective range of 13,986 feet. This includes a factor of safety of 1.3.

As can be seen in Figure 5-3, all of the battery packs considered meet the minimum range requirement. Therefore, the fuel efficiency of each pack must be considered. Since a prime design objective is low weight, the lightest battery pack which provides the necessary range would be ideal.

With these considerations in mind, Alpha Design selected the Top Flight 10-4 propeller, Gates 650-SCR 12-cell 650 mah battery pack and the Cobalt Astro-15 electric motor to provide good performance over the entire flight envelope.
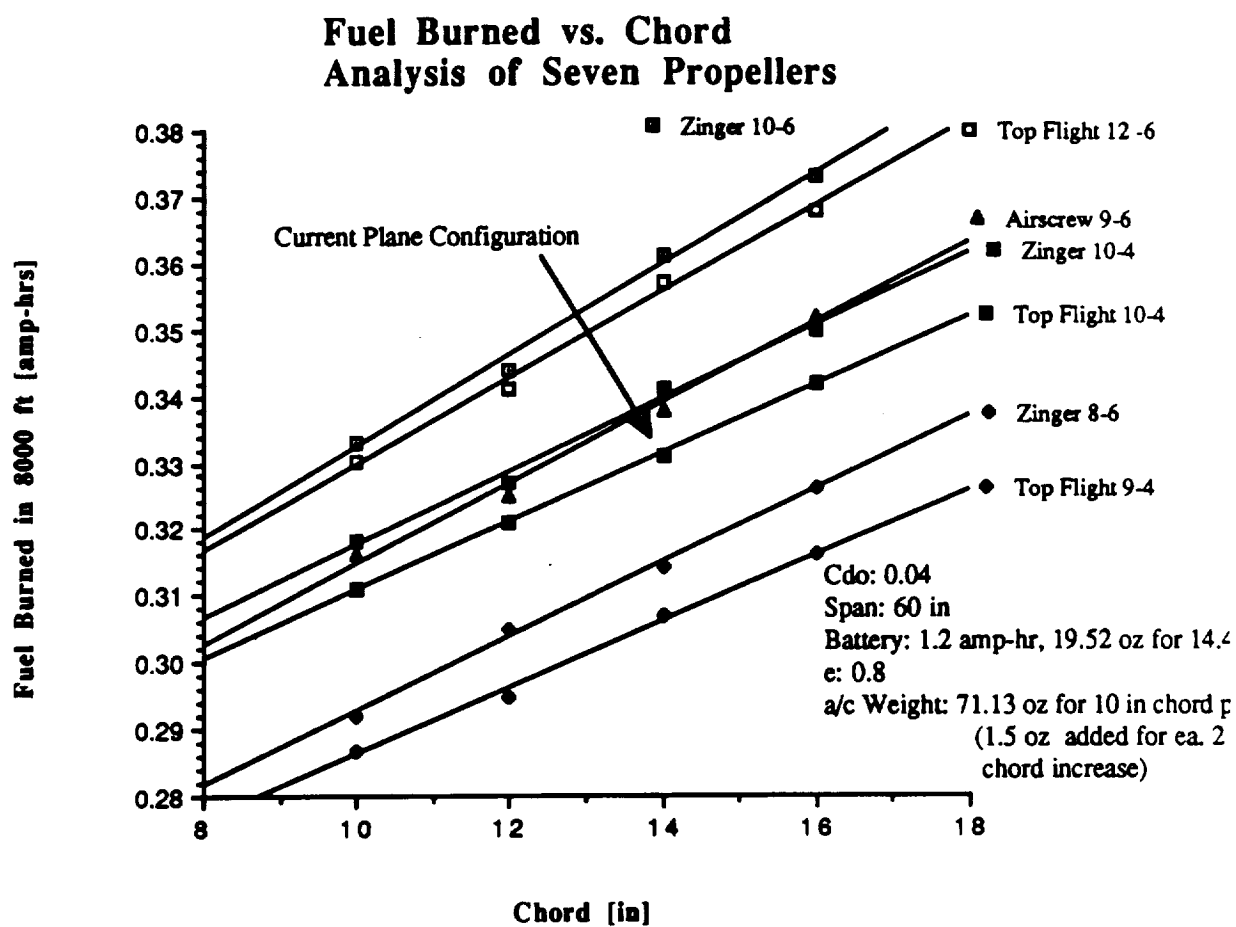
Fuel Burned vs. Chord
Analysis of Seven Propellers

Figure 5.1 Propeller Analysis.

Propeller efficiency vs. Advance Ratio
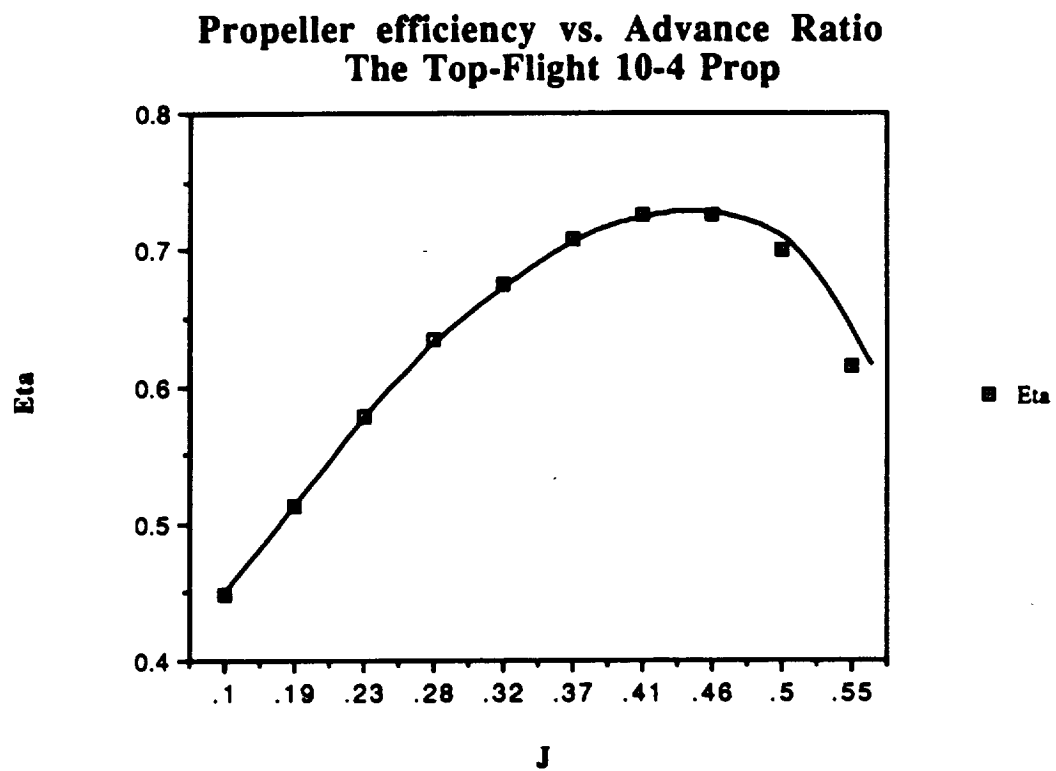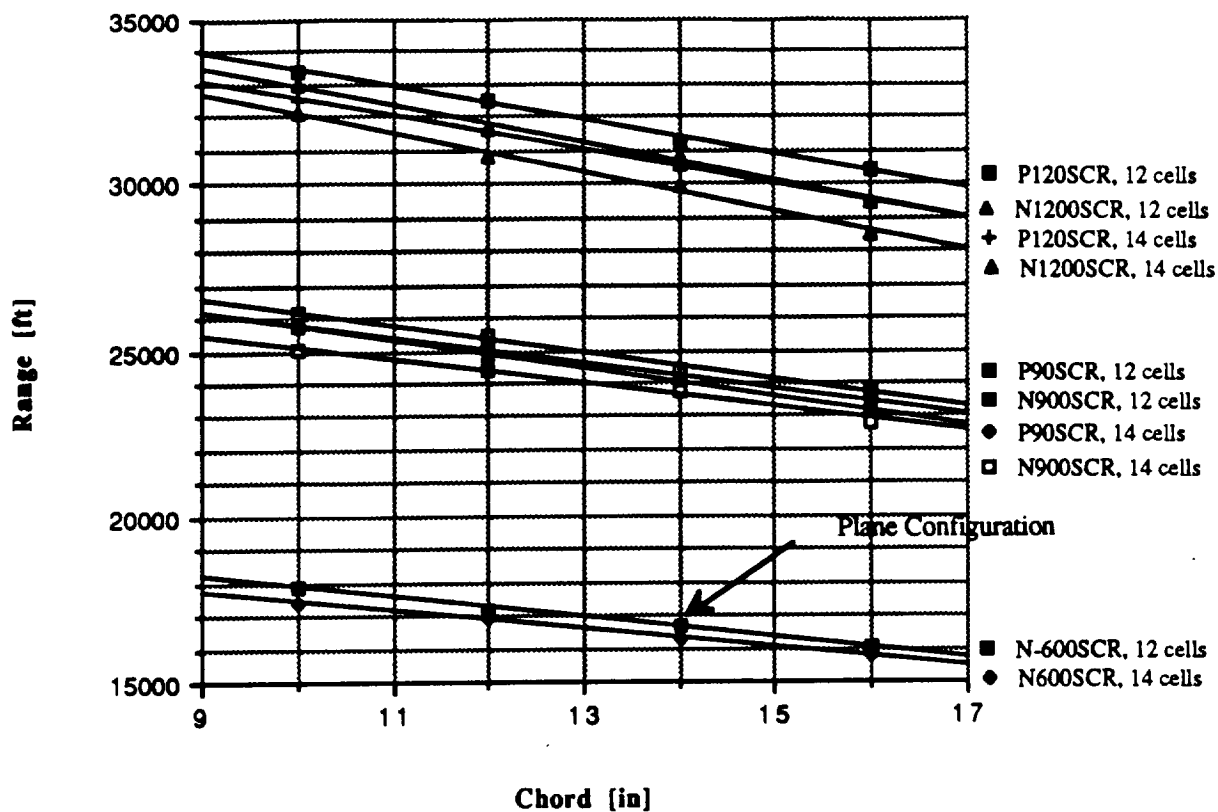**The Top-Flight 10-4 Prop**

Figure 5.2 Propeller Efficiency as a Function of Advance Ratio.

Range vs. Chord for various Battery Packs

Fixed for this Analysis:

V_{cruise} = 32 ft/s

C_{d_0} = 0.04

Span = 60 inches

Propeller = Top Flight 10-4

A/C Weight = 43.71 oz. without wing or battery
(Battery weight and 1.8 oz/ft^2 of wing
weight added.)

Figure 5.3 Battery Pack Selection.

6.0 WEIGHT ESTIMATION

6.1 Weight Analysis

When designing an aircraft, the weight is an important variable that can not be ignored. The weight of the aircraft adversely affects most of the plane's performance; therefore, it is essential to keep the weight as low as possible. From the start, designing the *Behemoth Apteryx* was greatly influenced by the overall weight. The importance of minimizing weight and the optimizing location of the center of gravity was especially critical for our design becasue our small wingspan gave us a high wing loading of 10.5 oz/ft^2. Initial weight estimates were made based on models from previous years. The weight and weight percentages of other RPV's were looked up, and from these values a heaviest weight was calculated for our aircraft.

The first step in determining our weights was to figure out the weights of each componenet of the aircraft. The wing weight was based on a structure presented by Mr. Joe Mergen,and determined based on the difference in planform area between his design and Alpha Design's. Refined weight estimates were performed as the actual weight of some of the components -- servos, controllers, cables, motor, battery, landing gear, and propeller -- became known. Eventually, an initial detailed drawing of the internal structural layout of the RPV was drawn up, and from this the volume of each individual structural component was determined. Combining the volume of the components with their material and density, the weight of the fuselage, empennage, and wing were calculated. At this point, a more refined analysis of the total weight of the aircraft was possible.

Building Materials	E (psi)	density, ρ (lbf/in^3)	stress, σ_{xx} (psi)
Balsa Wood	65,000	.0058	400
Spruce Wood	1,380,000	.0160	6,200
Plywood	2,010,000	.0231	2,500

Table 6.1 Properties of Building Materials.

Having determined the weight of each of the individual components, an initial estimate on the center of gravity could be made. Using the rough approximations for the (x,y) coordinates of each of the individual components, the location of the center of gravity was calculated. As the component weights and internal structural layout became more

exact, a better approximation of the center of gravity location could be made. The design was driven by simplicity and functionality. The passenger compartment and servos have been placed along the bottom surface of the fuselage in order to lower the center of gravity in order to increase stability. The one design parameter that had potential to alter the center of gravity position was the placement of the engine battery pack. For this reason, the battery pack has been designed to allow travel along the longitudinal axis in order to offset any shifts within the passenger compartment. This set up was to accomodate our desired range ($0.28c \leq$ c.g. $\leq 0.45c$) for the center of gravity will provide for a safer aircraft that is easier to control.

COMPONENT	WEIGHT oz	WEIGHT %	X-COORDINATE	Y-COORDINATE
ENGINE	8.76	14.21	2.00	2.50
BATTERY	12.67	20.54	6.00	3.00
WING	9.00	14.59	10.50	5.50
FUSELAGE	11.6	18.80	17.40	2.80
EMPENNAGE	4.15	6.73	44.50	4.00
MAIN GEAR	4.15	5.54	6.00	-2.50
REAR GEAR	0.5	.81	44.00	-1.50
AVIONICS	6.58	10.67	11.50	2.00
PROPELLER	0.50	.81	0.00	2.50
PAYLOAD	4.50	7.29	27.50	0.75

Table 6.2 Center of Gravity Positions.

The estimated weight of the aircraft was constantly used to update the required thrust for take off as well as wing sizing. The internal positioning of the aircraft's batteries and servos affected the center of gravity, which in turn affected the static and directional stability of the aircraft. The constant effort to reduce the overall weight of the aircraft demanded that the exact location of the c.g. and the planform area of the wing be frequently evaluated in order to determine the necessary changes in the internal layout. The actual weight of the *Behemoth Apteryx* coincides with the predicted design values. Current efforts are directed at placing the c.g. at the optimum position, which is at .33c.

7.0 STRUCTURAL DESIGN

7.1 V-n Diagram

The *Behemoth Apteryx* will operate in a flight envelope shown in Figure 7.1. The limit load factor for our aircraft is 1.8, and the ultimate load factor is 2.5. The largest load factor that is expected in normal flight operations is 1.4 during a banked turn. This value is given a safety factor of 1.3 to allow for quick obstacle clearance that will require pulling slightly higher g's. In Figure 7.1, the cruise velocity is indicated at 32 ft/s and the sound barrier at 35 ft/s. The aircraft is designed for a maximum velocity of 52 ft/s. As indicated in the figure, the normal cruise condition for our aircraft is well within the envelope. The negative limit load factor is -.5 which is more than adequate for unintentional pilot induced maneuvers, provided they are not violent.

7.2 Structural Components

The initial approach in the design procedure of the wing, fuselage and empennage of the *Behemoth Apteryx* was based on empirical information, that is, we examined successful designs of previous years and used their techniques for our custom configuration. This empirical data, which includes the database, design room models, and the advice of Mr. Joe Mergen, turned out to be an invaluable source of information.

Due to the smooth, even surface of the runways in Aeroworld, there are no significant ground loads during taxi. The greatest stresses during takeoff will occur at the leading edge of the wing when the aircraft is rotated and liftoff is achieved. During flight, the greatest stresses will occur in the main fuselage and the wing when the aircraft is banked for a turn. One of the most important tasks, therefore, is to locate the elements in these areas where the stresses are largest and increase the strength by either using stronger materials or increasing the size of the elements. However, since the greatest show-stopper for our design is being able to generate enough lift at takeoff, keeping the weight low is essential.

7.2.1 Wing

The wing design for the *Behemoth Apteryx* is shown in Figure 7.2. This design was based on two driving factors: simplicity and low weight. Because cost is an important

parameter in this design project, keeping the wing design simple will decrease production and maintenance costs and the importance of keeping the structural weight low is obvious.

The wing was originally designed to span 5 feet, with a chord of 12 inches. We used SWIFTOS, a finite element program, to analyze the performance of the wing for a load factor of 2, which is well beyond the expected normal operating conditions. The placement of the main spar was optimum at 30% chord and the sizing of the main spar was originally 1/16 inches. Built-up leading and trailing edge spars were chosen after examining past designs. Ribs were sized at 1/16 inches, and the spar caps were sized at 3/16 x 1/16 inches. These figures were adequate for our design, but there were two factors that changed our final design. First, there was an uncertainty in the use of SWIFTOS due to the difficulty in modeling the skin of the wing (Monokote). Second, it was later decided to increase the chord of the wing to 14 inches in order to generate the needed lift. This resulted in increasing the gauge of the ribs, main spar, and the spar caps. The wing will be constructed using balsa wood and shrinkable plastic skin covering.

It is expected that failure in the wing structure will occur in the main spar at the root of the wing due to axial stresses resulting from a large bending moment. The final wing design was re-modeled using SWIFTOS, and none of the wing element stresses exceeded the maximum allowable stresses for balsa wood, with a comfortable margin of safety. It was originally decided that the root of the wing would be reinforced by fiberglass, but after further analysis, the adjustments required to facilitate the attachment of the fiberglass would be too complicated and add too much weight to the structure. The wing design should be adequate if our limit load factor is not exceeded. This conclusion is based on our beam bending analysis which was done assuming main spar caps carry all bending and using a factor of safety of 4.

The wing design was greatly simplified by the fact that there was no sweep, taper, or folding necessary. The only complications that could arise would be in constructing the correct dihedral angle and incorporating ailerons into the structure. Alpha Design is also investigating the use of winglets to increase the efficiency of the wing, but these would be only an addition to a derivative aircraft in order to accomodate the necessary performance improvements needed with the increased loads of additional passengers. The wing will be connected to the fuselage by two screws at the root of the trailing edge and pegged at the main spar as shown in Figure 7.3.

7.2.2 Fuselage

We realized that a complete analysis to produce the optimum fuselage design would be a monolithic task, so we relied heavily on empirical data for our design. We concluded that a simple truss structure would provide the necessary strength without increasing the weight. The configuration of the fuselage is shown in Figure 7.4. This design will allow for just enough volume to house the necessary components: cargo, motor, receiver, servos and batteries. The box shape was picked to keep the design simple. The greatest difficulty arose in the sizing and material selection for the fuselage elements. Ideally, the fuselage would be constructed entirely of balsa wood. However, after analyzing the problem using a simple rigid body model, stresses were located under the wing that came too close to exceeding the allowable stresses for balsa wood. For these high-stress regions, harder wood (spruce) will have to be used. These elements are indicated by an "s" in Figure 7.4. Plywood will also have to be incorporated into the structure for geometric reasons. The motor will be need to be mounted on a sheet of strong wood; if spruce cannot be found in the necessary geometry, plywood, not as strong as and more dense than spruce, will be used in construction. This area is indicated by a "p" in Fig. 7.4.

7.2.3 Empennage and Landing Gear

The horizontal and vertical tails are flat plates constructed of balsa wood to keep the design simple and the weight down. The elevator is 1.5 inches long (chordwise) and is a symmetric airfoil shape. The rudder is notched to prevent obstruction when both the elevator and rudder are deflected. Figure 7.5 shows the empennage configuration.

The Apteryx has a tail-dragger landing gear configuration. For the main gear, we chose to use a $\frac{3}{32}"$ steel rod, bent to our specifications. This gear will be light-weight and will supply enough cushion to absorb some of the energy during landing impact (Figure 7.6). We will also be able to bend the main gear in case the angle needed to take-off is adjusted. The main gear will use 2 inch wheels and the tail-dragger, which is connected to the rudder, will use a 1 inch wheel.

7.3 Materials Selection

The materials that were considered for our design were woods, metals, composites, and plastics. Composites, although lightweight and strong, have availability and cost

problems. Metals are strong and readily available, but have weight problems. With our design goals and critical factors of weight and simplicity, we really had no other choice than to use as much balsa wood as possible, along with plastic skin covering. Wood is most readily available, easy to machine, and relatively lightweight. The only metal used in the structure will be the motor mount and landing gear.
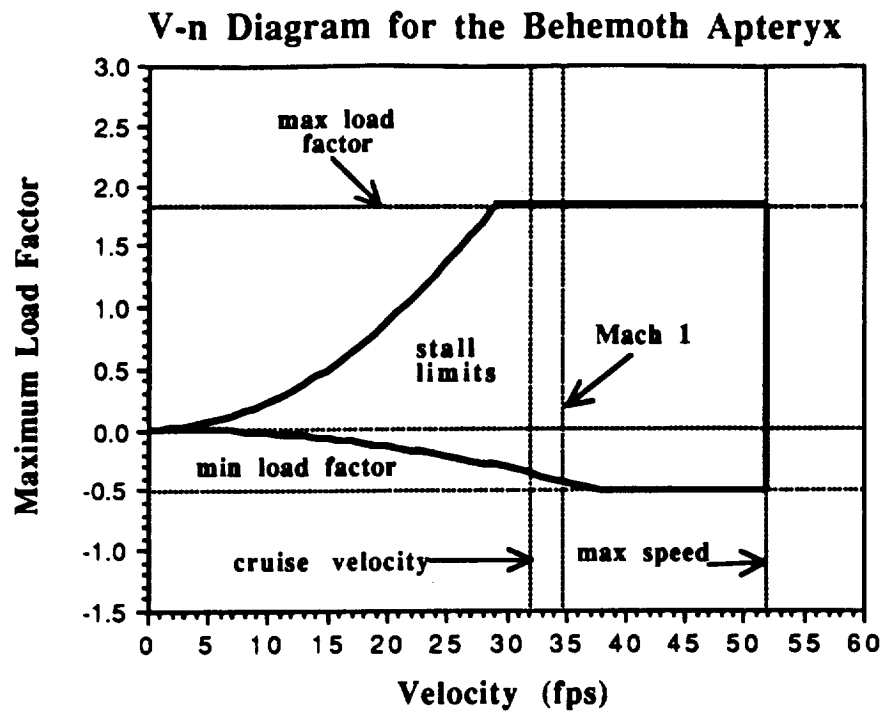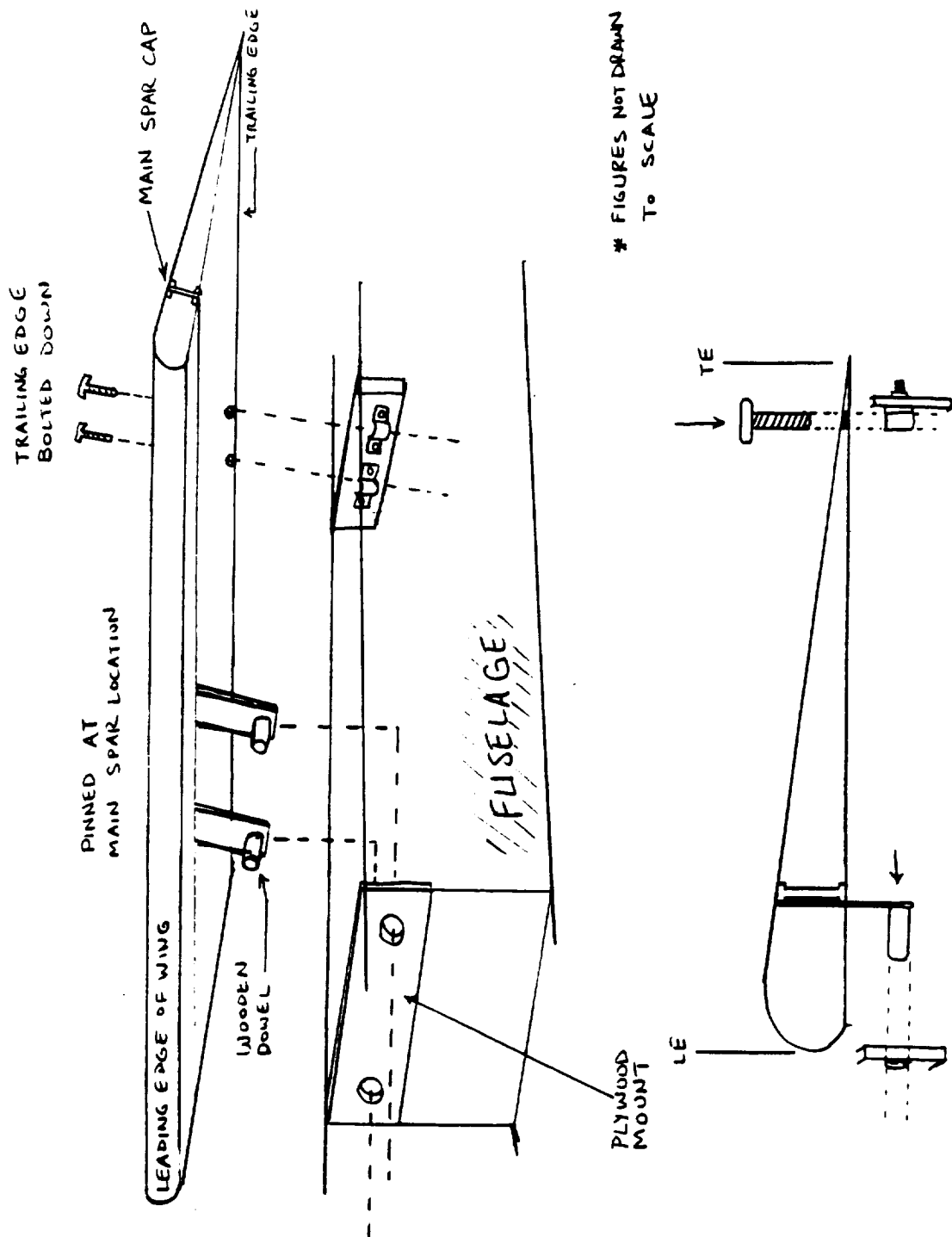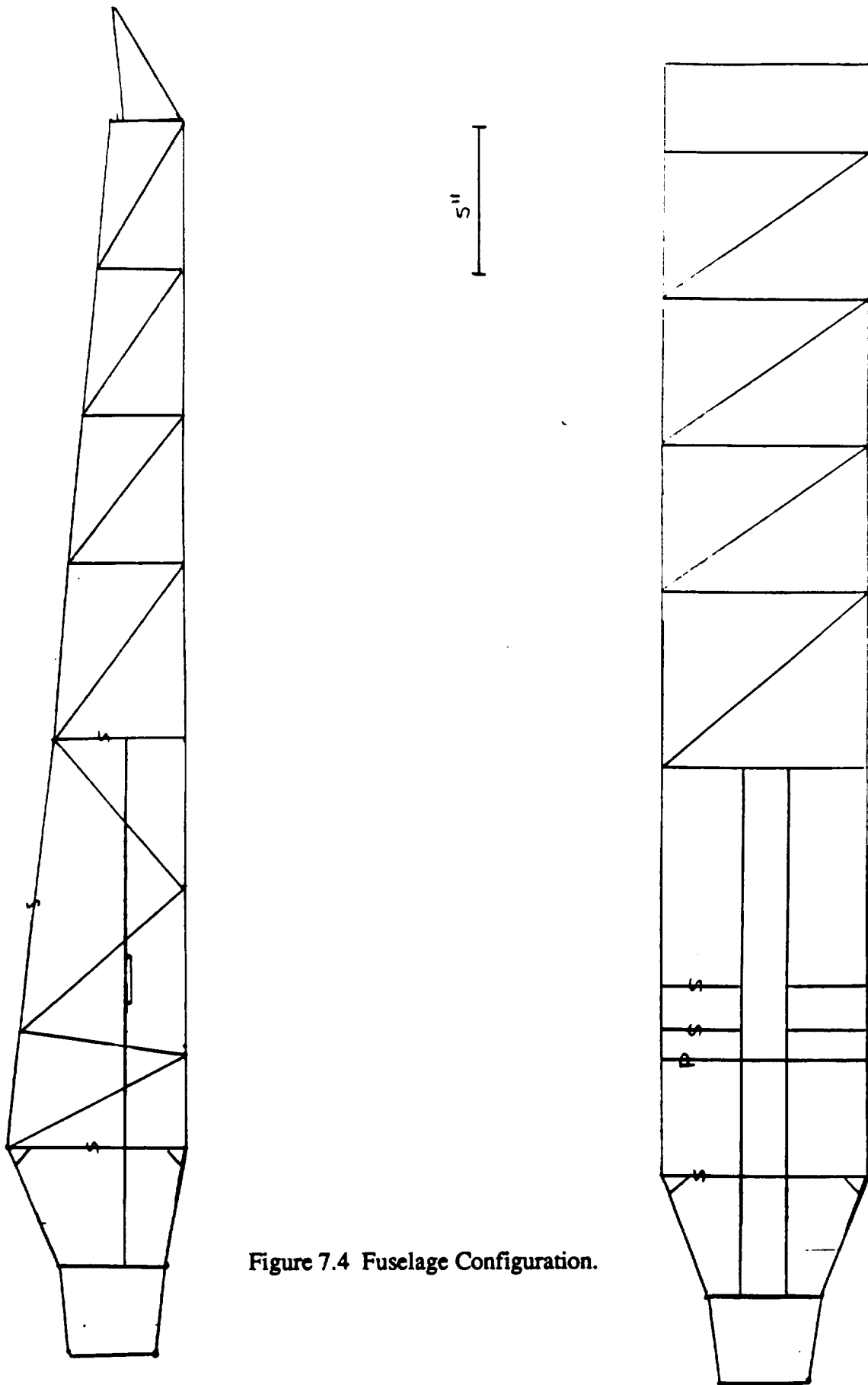
Figure 7.1 V-n diagram for Behemoth Apteryx.

ROOT

30"

18"

1.25"

14"

1"

$\frac{1}{8}$"

$\frac{3}{16}$" × $\frac{1}{8}$"

$\frac{1}{8}$"

LE

TE

LEFT WING

Figure 7.2 Wing Design.

Figure 7.3 Wing Mounting Design

5"

S = SPRUCE
P = PLYWOOD
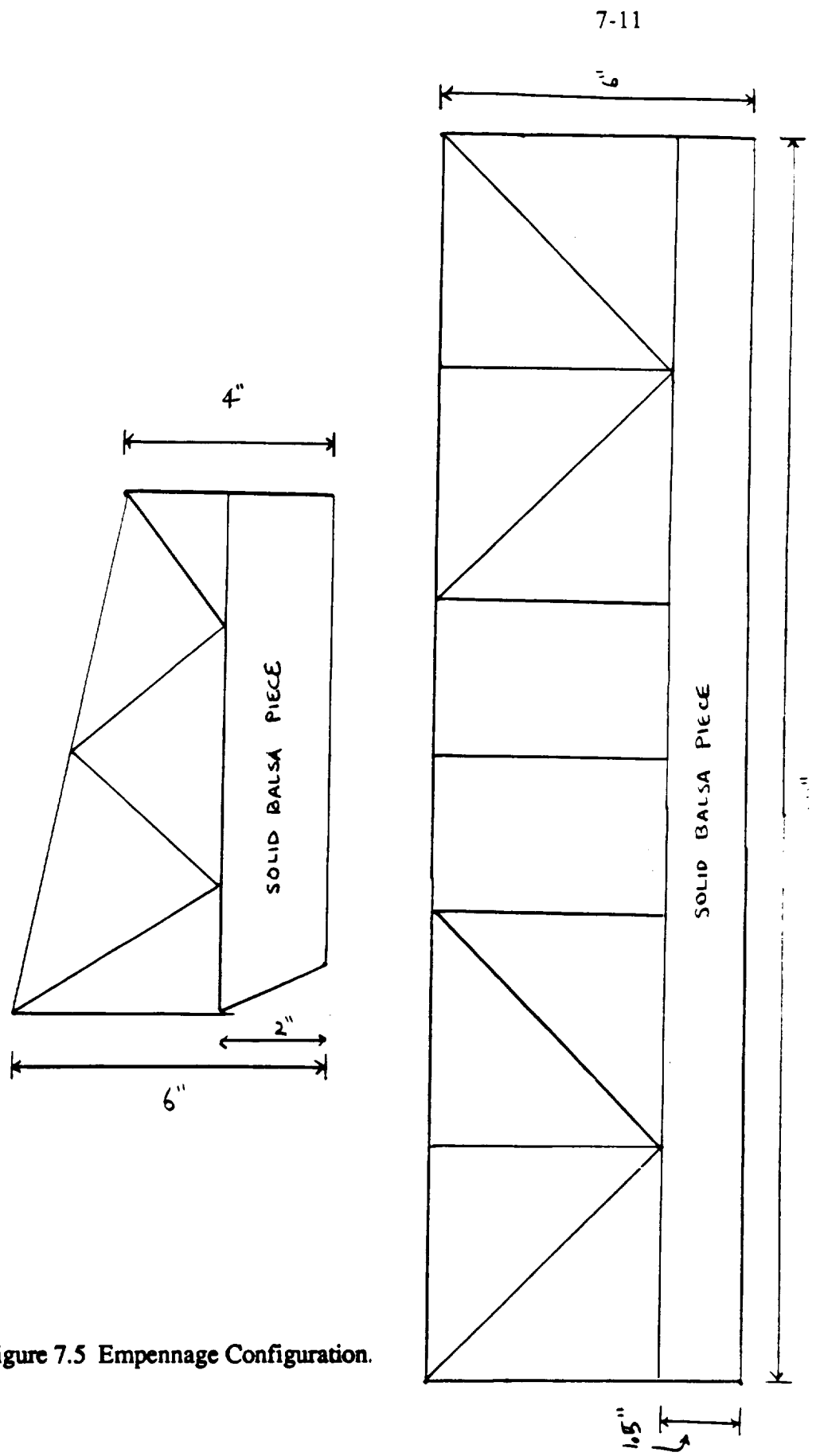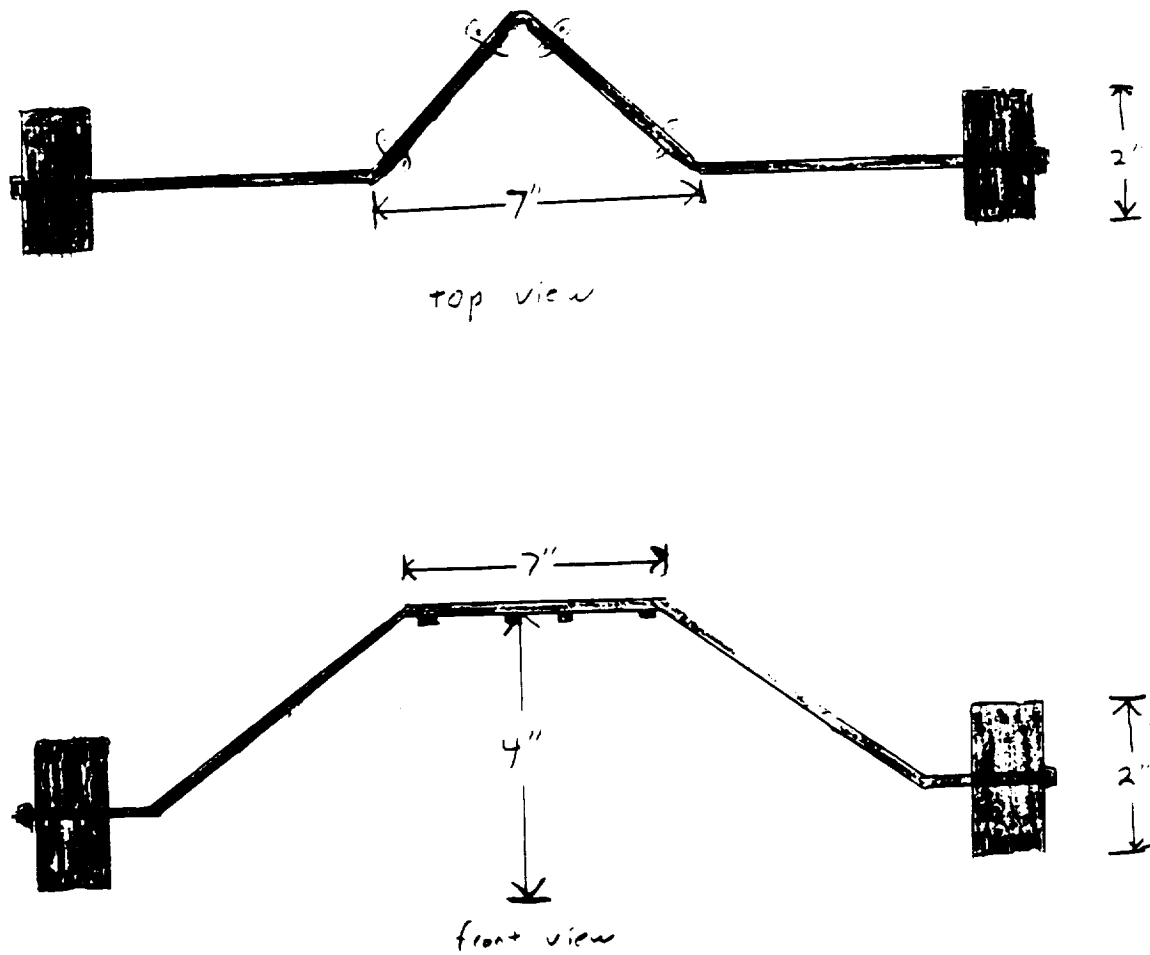* NO INDICATION = BALSA *

Figure 7.4 Fuselage Configuration.

Figure 7.5 Empennage Configuration.

top view

front view

* note: figure not to scale

Figure 7.6 Landing Gear Configuration.

8.0 PERFORMANCE ESTIMATES

8.1 Take Off & Landing Ground Roll

According to the mission definition presented by Alpha Design at the beginning of the semester, our aircraft must satisfy two self imposed requirements: its wingspan can be no greater than five feet, and it must be able to take off and land in fifty feet. Additionally, once the aircraft has successfully lifted off the ground, it would proceed to cruise at an altitude less than 35 feet and at a cruise speed of 32 feet per second. Alpha Design proposed to construct an aircraft that could fly 8000 feet (this distance includes loiter and redirect time as well as landing) and carry 50 passengers. The self imposed restrictions on the take off distance of the Behemoth Apteryx greatly affected the final design and performance of the aircraft.

The theoretical ground roll during take off was calculated using the following equation and TK! Solver Plus.

$$X_{gr} = \frac{1.44W^2}{g\rho S C_l \{ T - [D + \mu_r (W - L)]_{ave} \}}$$

This equation and program were used to probe the effects of various parameters and their impact on the take off distance. From these studies it was determined that a larger chord would be necessary in order to fulfill the requirements. During the the take off ground roll analysis, several things were assumed: the rolling friction coefficient, μ, was equal to 0.04, the atmospheric conditions were constant, and the denominator of the expression *(above)* could be accurately calculated at 70% take off speed. Of all the variables in the equation, the lift coefficient, the thrust, and the weight were found to have the most significant impact on take off ground roll(see figures 8.1 and 8.2). This equation calculated the optimum ground roll, and other external factors could adversely affect the optimum ground roll. In order to successfully take off within the specified distance, the design of the aircraft had to be modified such that it had more lifting surface.

Part of the take off analysis involved determining the landing distance of the aircraft. Using a variation of the take off distance equation *(below)*, the theoretical landing distance was determined. During the the landing ground roll analysis, several things were assumed: the rolling friction coefficient, μ, was equal to 0.04, the atmospheric conditions were constant, and the denominator of the expression could be accurately calculated at 70% take off speed.However, it was noted that the predicted values were roughly an order of magnitude larger than the values obtained over the past years from other remotely piloted vehicles (RPV's). A possible explanation for this is the use of an incorrect value for the

rolling friction coefficient in the take off and landing equations. However, the effect of varying μ on the landing distance was determined (see Figure 8.3) and it was realized that this factor alone could not account for the discrepancy.

$$X_{gr} = \frac{1.69W^2}{g\rho SC_l \ [D + \mu_r \ [W - L]]_{ave}}$$

According to the theoretical calculations, the landing distance needed to bring the aircraft to a full stop would be approximately 180 feet; however, the empirical data collected from previous RPV's indicate landing distances of roughly 30-60 feet. While calculating the landing distance of our aircraft, the effects of thrust reversal and wheel braking were documented. Assuming a friction coefficient of $\mu=.40$ (an effect of wheel braking) and 40% thrust reversal, the RPV would be able to stop within the distance specified. As part of the landing distance analysis, the effects of the friction coefficient and thrust reversal were also investigated (see Figures 8.3).

8.2   Range & Endurance

The analysis of the range and endurance of the *Behemoth Apteryx* considered the impact of three variables: aircraft weight, lifting surface area, and the battery capacity. The effect of weight was charted as refinements were made in the design that in turn lowered the weight of the aircraft. The wing area is not a parameter that was varied with the intention of altering the design because of its impact on other performance figures; however, the battery capacity was lowered from our initial guesses of 1200 mah down to 650 mah in efforts to reduce the weight and to more accurately realize our design goals. The theoretical range and endurance of the aircraft and their sensitivities to other parameters were calculated using TK!Solver Plus. Using this program, the range and endurance were calculated by setting the rate of climb to zero and then the range and endurance were plotted against the cruise velocity (see figures 8.4 and 8.5). When evaluating the maximum and minimum values of range and endurance and where they occur, a few interesting facts are observed. Electric powered RPV's apparently are not governed by the same rules that apply to gasoline propeller driven aircraft. When examining the data two questions come to mind: why doesn't the maximum endurance occur at $\frac{Cl^{3/2}}{Cd_{max}}$ and why doesn't the maximum range occur at $\frac{Cl}{Cd_{max}}$? These relationships apply to gas powered propeller driven aircraft because the weight of the aircraft changes as the flight proceeds; however, with electric powered RPV's there is no weight change. After examining the data (see Table 8.1) it is clear that the remotely piloted vehicle will be flying at nearly the optimum

speed for endurance and rate of climb. The range of this aircraft does not peak until a flight speed of 44 feet/sec, but it can easily obtain the desired range of 8000 feet at any flight speed. All of the calculated values for range and endurance were done assuming steady level flight. Since the zero lift drag coefficient rises in turning flight, the actual range will decrease approximately 15-20%. It should be noted that the range and endurance were calculated assuming steady level flight, and these figures were determined by theoretically depleting the fuel supply. These range figures include loiter, redirect, and landing fuel. Therefore, approximately 20% of the range should be deducted for the adverse effects of turning flight and a small additional percentage should be deducted for amount of fuel desired left over at the end of the flight.

data based on .600 milliamp hours after take off	FLIGHT TIME (optimum conditions)	RANGE (optimum conditions)	RATE OF CLIMB
MINIMUM based on steady, level flight	8.20 minutes @ V = 25 fps	12306.7 feet @ V = 25 fps	5.72 fps @ V = 25 fps @ 14.4 Volts
MAXIMUM based on steady, level flight	8.47 minutes @ V = 31 fps	19705.7 feet @ V = 44 fps	6.18 fps @ V = 31 fps @ 14.4 Volts
CRUISE based on steady, level flight	8.46 minutes @ V = 32 fps	16236.8 feet @ V = 32 fps	6.17 fps @ V = 32 fps @ 14.4 Volts

Table 8.1 Range and Endurance Predictions.

8.3 Power Required, Power Available, and Rate of Climb

During the cruise portion of the flight, Alpha Design's remotely piloted vehicle will require 15.04 Watts power in order to maintain cruise, and this is achieved with a motor current draw of 4.17 amps. After ascertaining the power required at cruise, the next logical step is to calculate the power available and the rate of climb at various speeds. Power available is a critical figure in the performance of this aircraft. The rate of climb is a very important performance figure because this aircraft will be operating within the close confines of the Loftus Sports facility. The aircraft must be able to obtain cruise altitude quickly so that it may initiate the preset course. The required and available power were determined using TK Solver to establish the sensitivity of the aircraft to flight speed (see Figure 8.7). Once the data was plotted, a simple evaluation of the graph established the

absolute maximum and minimum flight speeds. The rate of climb is a function of the available and required power as well as the vehicle's weight. Using the equations below, the rate of climb was then calculated and graphed (see Figure 8.6).

$$P_{required} = D \times V = \frac{W}{\frac{C_l}{C_d}} V_\infty = \sqrt{\frac{2W^3 C_d^2}{\rho_\infty S C_l^3}} \qquad \text{Rate of Climb} = \frac{P_{available} - P_{required}}{\text{Weight}}$$

Part of the power required analysis involved determining the performance of the aircraft in the event of a complete power failure. If the aircraft were to lose engine power, then the pilot would be forced to glide the plane to a landing. Using the below equations

$$\gamma_{min} = \arctan\left(\frac{1}{\left(\frac{L}{D}\right)_{max}}\right) \qquad X_{best\ range} = \left(\frac{L}{D}\right)_{max}) * altitude \qquad \left(\frac{L}{D}\right)_{aircraft\ max} = 10.8$$

the minimum glide angle, γ, and the best glide distance, $X_{best\ range}$, were determined to be $\gamma = 5.28°$ and $X_{best\ range} = 270.75$ feet. This means that in the event that the aircraft were to run of engine battery power, the RPV still could maneuver and land safely in Aeroworld.
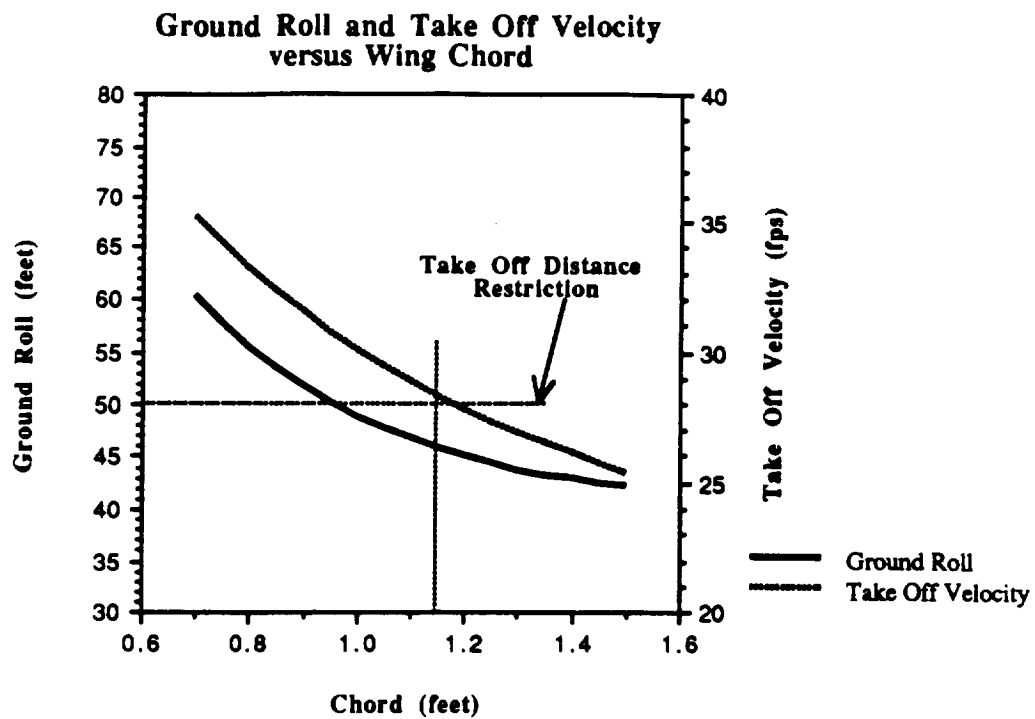
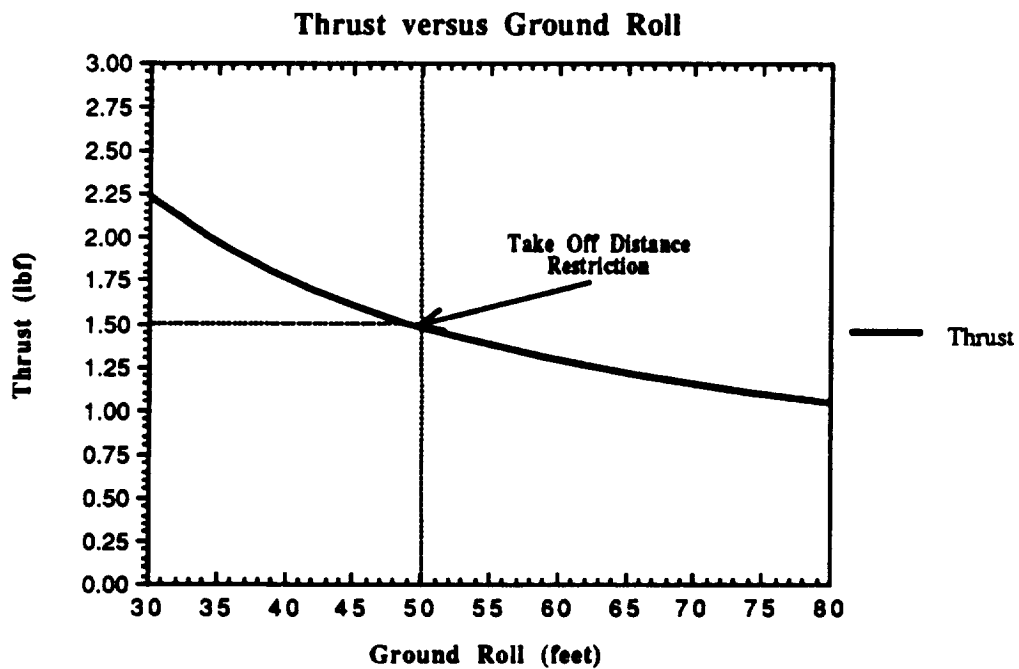Figure 8.1 Ground Roll & Take Off versus Wing Chord
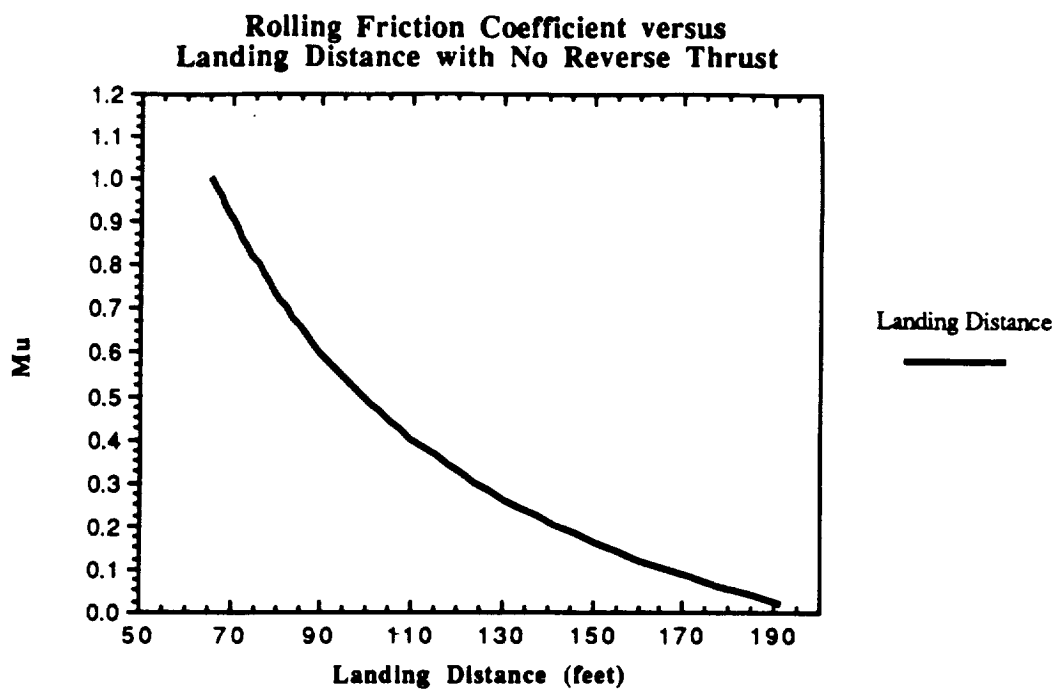

Figure 8.2 Thrust versus Ground Roll

Rolling Friction Coefficient versus
Landing Distance with No Reverse Thrust

Landing Distance

Figure 8.3 Landing Distance vs. Rolling Friction Coefficient
No Reverse Thrust

Range versus Velocity
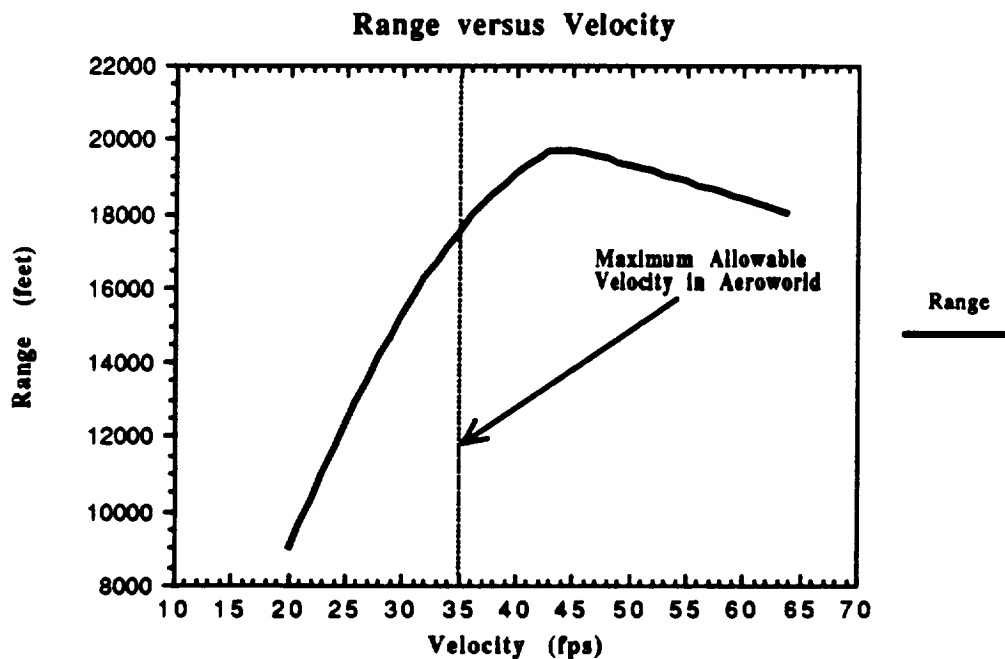
Range

Figure 8.4 Range vs. Velocity

Endurance versus Velocity
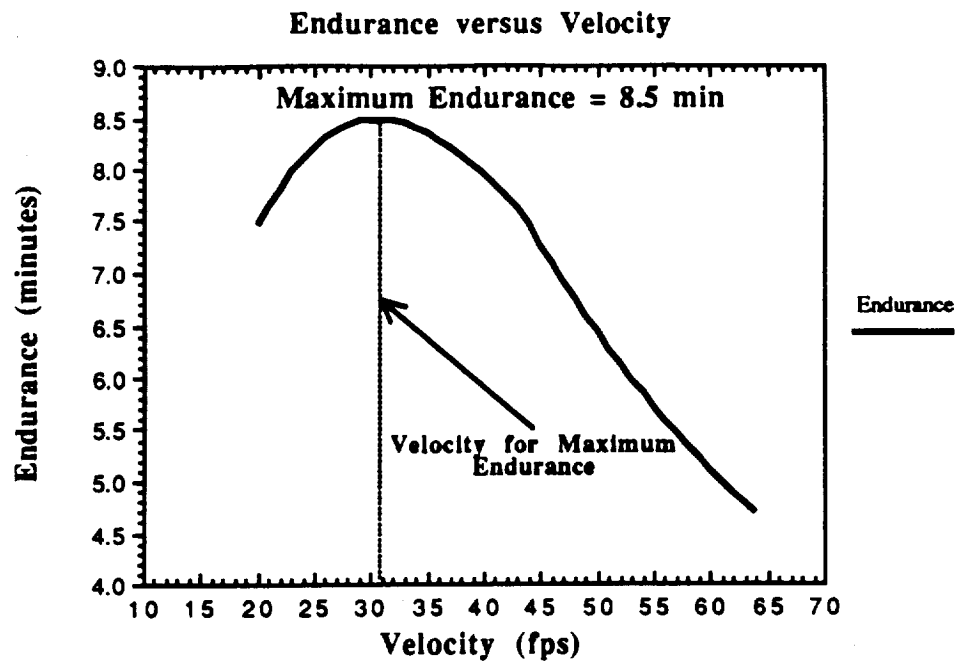
Figure 8.5 Endurance vs. Velocity

Rate of Climb versus Velocity
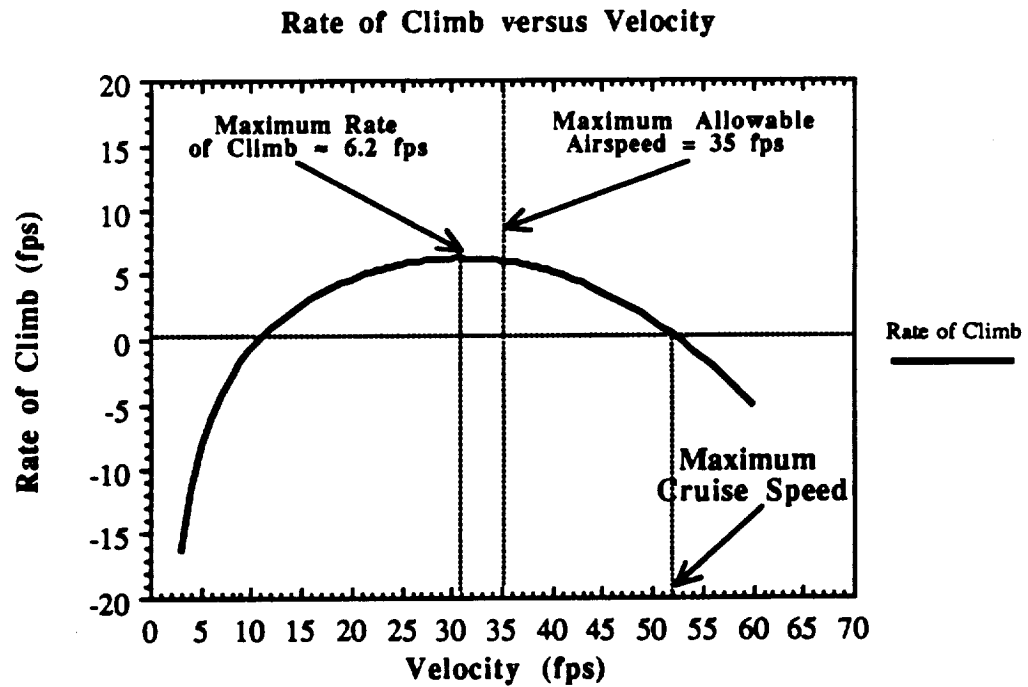
Figure 8.6 Rate of Climb vs. Velocity

**Power Available & Required
versus Velocity**
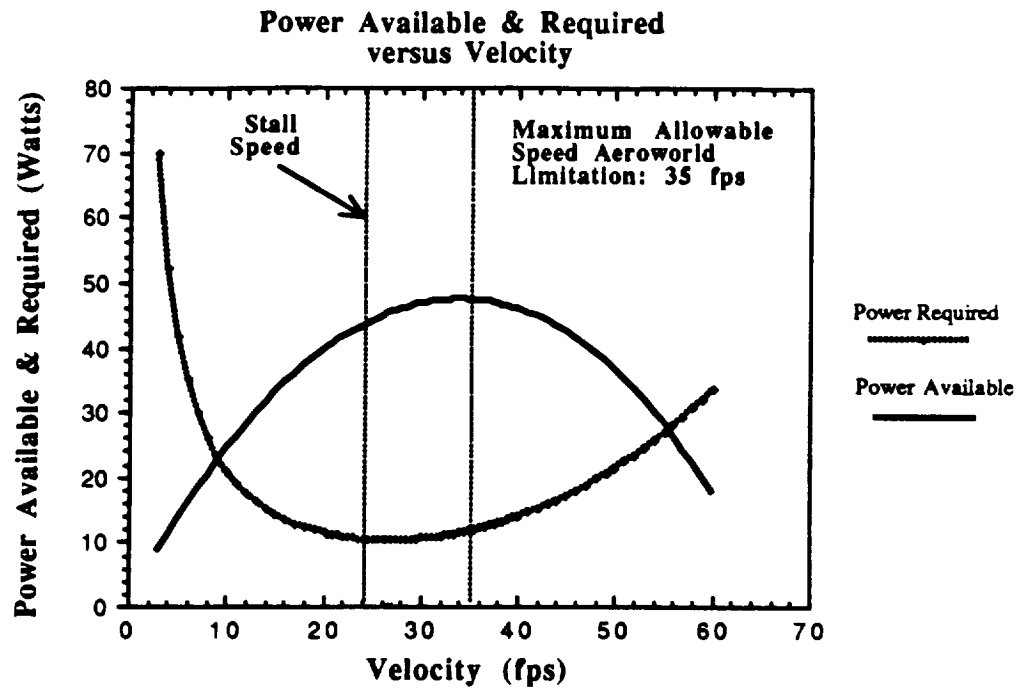
Figure 8.7 Power Available & Required versus Velocity

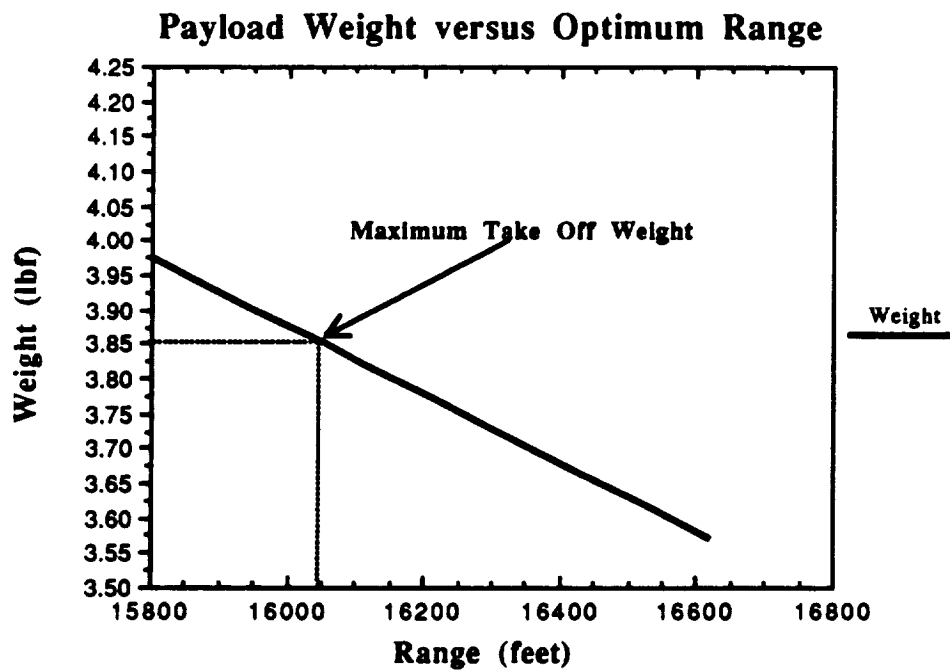**Payload Weight versus Optimum Range**

Figure 8.8 Payload vs. Range

9.0 ECONOMIC ANALYSIS

This section will address the economic justifications for our aircraft design. This includes determining ticket prices that will meet or beat our competitors fares on the routes that will be served by our commercial transport. In order to set ticket prices, we first needed to determine the fixed and variable costs associated with aircraft production, the number of aircraft that would be produced, the total number of flights flown per day, and the total daily passenger load.

9.1 Cost Estimation

9.1.1 Fixed Costs

The first task was to identify all the fixed costs and operating and maintenance costs for a single 50-passenger aircraft. The fixed costs include the costs of material and labor during production, while the operating and maintenance costs include fuel, maintenance, and the crews' salaries. The complete cost breakdown is shown in Table 9.1. As can be seen in Table 9.1 under Fixed Costs, the propulsion and control systems make up the bulk of the total fixed cost. Since these systems will be purchased from a subcontractor at a set price, Alpha Group will not significantly be able to reduce the total fixed cost. This is a consequence of the fact that these systems (the geared motor, speed controller, 4 channel radio, receiver, and 4 servos) constitute 62% of the $211,400 total production cost for each aircraft. The prices for these systems were obtained form Hobbyland in South Bend, Indiana. The fixed costs that Alpha Design will have some control over are the costs of material and labor required to build a safe, efficient aircraft. The labor costs have been estimated at $15,000 per aircraft. This was determined for 150 construction man-hours per aircraft at a rate of $100 per construction man-hour. The cost of monokote will run approximately $8000 per aircraft and the estimated cost of other materials (balsa wood, hardwood, and landing gear) is $40,000 per aircraft. These labor and material costs only constitute 30% of the total production cost of each aircraft. Therefore, even if these cost estimations are inaccurate, the total cost to produce our commercial transport will not change significantly if more or less man-hours and wooden materials are needed to complete construction.

PRODUCTION COSTS

Fixed Costs

Astro Cobalt 15 geared motor	$44,800 / aircraft
4 channel radio, receiver, & 4 servos	$56,000 / aircraft
Speed Controller	$28,000 / aircraft
Propeller	$ 1,200 / aircraft
12 cell Nicad battery packs	$18,400 / aircraft
Monokote (3 rolls)	$ 8,000 / aircraft
Other materials (balsa wood, glue, landing gear)	$40,000 / aircraft
150 hours of labor	$15,000 / aircraft

Total Fixed Cost   **$211,400 / aircraft**

Delivery Price   **$325,000 / aircraft**

Operating & Maintenance Costs

Fuel	$60-$120 / milli-amp hour
Maintenance	$500 / flight
Crew of 2 / flight	$200 / flight

Table 9.1 Cost Breakdown

9.1.2 Operating and Maintenance Costs

Estimating the operating and maintenance costs was somewhat more difficult. This is because the price of fuel can fluctuate between $60 and $120 per milli-amp hour. Also, there was difficulty in determining how many milli-amp hours would be burned per foot by the aircraft while in flight. A good estimate of 0.04 milli-amp hr / ft is used in determining fuel costs. This number is a slightly pessimistic value to allow for error in propulsion calculations and aircraft design changes and to show the airlines that they will still realize a large profit if this value were exact. The value may actually turn out to be as low as 0.035 milli-amp hr / ft. It should be noted this fuel burn-rate value is very crucial to the economic analysis because the fuel costs are by far the largest single cost for the airline. Maintenance time will only require 1 minute RWT (30 minutes AWT) and one person because the batteries will be conveniently located beneath the wing in the top half of the fuselage. This translates into a cost of $500 per flight. Each aircraft will require a two-person crew. For this analysis, we assumed that the airline purchasing our aircraft will pay each crew member $100 per flight. This $100 per flight is more or less just an average dollar figure as pilots will alternate flying between short and long routes.

9.2   Production

The next step was to determine the total number of aircraft that will be produced for our proposed market. The aircraft produced will serve the entire northern hemisphere of Aeroworld, providing daily service to-and-from cities A, B, F, G, H, I, J, K, J, L, M, and N. A sketch of our market's route system is shown in Figure 9.1. To determine how many aircraft are needed to service this system, a daily time schedule was systematically developed for one plane at a time until the entire route system was covered. The time table was developed using an average cruising speed of 32 ft/s per aircraft per flight and allows each aircraft at least 30 minutes at the gate between flights for refueling, passenger deplaning, and passenger loading. By developing the table, we were able to determine that we need to produce 38 aircraft to fulfill our market goal in Aeroworld. Once the airline purchases the 38 aircraft, they can use a similar time table to schedule the 584 full flights per day necessary to carry 29,200 paying passengers. By calculating route distances, it can be shown that the entire fleet will fly a total of 1,902,500 feet per day over Aeroworld. Multiplying this figure by 50 passengers per flight translates into 95,125,000 revenue passenger-feet per day.

9.3   Profit

9.3.1   Profit Predicted for Alpha Design

Each aircraft will be delivered to the airline at a price of $325,000 per aircraft. For a fleet of 38 50-passenger aircraft, the total delivery price is $12,350,000. Since the cost to produce 38 aircraft will be $8,033,200, Alpha Design will realize a profit of $4,316,800 on the sale of the aircraft.

9.3.2   Profit Analysis for Purchasing Airline

Knowing this information, a formula was developed to determine how many days it would take for the airline to break-even for various ticket prices. In this analysis the ticket prices are based on dollars per foot. The following formula was used:

$$\text{Days} = \frac{\$12300000}{(95125000\text{ft})(x\frac{\text{dollars}}{\text{ft}}) - (.04\frac{\text{mahr}}{\text{ft}})(1902500\text{ft})(z\frac{\text{dollars}}{\text{mahr}}) - 584(\$200)}$$

In this formula and on the corresponding plot **Days** represents the number of days to break-even, hence pay for the $12,350,000 delivery price of the fleet of 38 aircraft. x represents a set ticket price in dollars per foot and z represents the price of fuel which can vary from $60 to $120 per milli-amp hour. Fig. 9.2 shows a graph of the formula for five different fuel costs for ticket prices up to $0.20/ft. If the plots were extended, it would become apparent that the airline would lose money and never break-even at certain fuel costs and ticket prices. This would occur because the airline would not make enough revenue each day to cover the daily fuel costs. For example, if the ticket price was set at $.09/ft and the fuel cost was $120 per milli-amp hour the airline would gain $8,561,250 per day from ticket revenue, but the fuel costs would be $9,132,000 per day. As can be seen, the cost of the fuel is by far the largest cost. As a matter of fact, in some scenarios the cost of the fuel burned each day is more than the $12,350,000 delivery price of 38 aircraft. Fuel efficiency therefore was an important driving factor for the aircraft design. Although our design produces more drag and is less fuel efficient than similar aircraft configurations with a longer wingspan and smaller chord, we have determined that the slight additional fuel cost will remain lower than the cost of maintaining a hydraulic system in a folding wing with a wingspan greater than 5 feet. The above formula and Figure 9.2 will be extremely useful to the airline in setting ticket prices as fuel costs fluctuate.

Figure 9.3 shows how much profit can be realized once the fleet of 38 aircraft have been paid off through ticket revenues. Hence, Figure 9.3 may be used to set ticket prices once Figure 9.2 is no longer applicable.

9.4   Market Competition

This economic analysis has shown that the airline can compete and in fact beat the fares of the average train and ship. The average train fare is $.125/ft + $50 flat rate and the average ship fare is $.16/ft + $65 flat rate. In all possible scenarios this airline can easily beat the ticket prices of its competitors and still make a highly respectable profit. Even if the airline sets its ticket price at $.125/ft (equal to the train fare minus the flat rate) and the fuel costs are $120 per milli-amp hour, the airline would still make a profit of $2.3 million per day. In a year, this translates into a profit of $840 million on ticket revenues $4.34 billion.
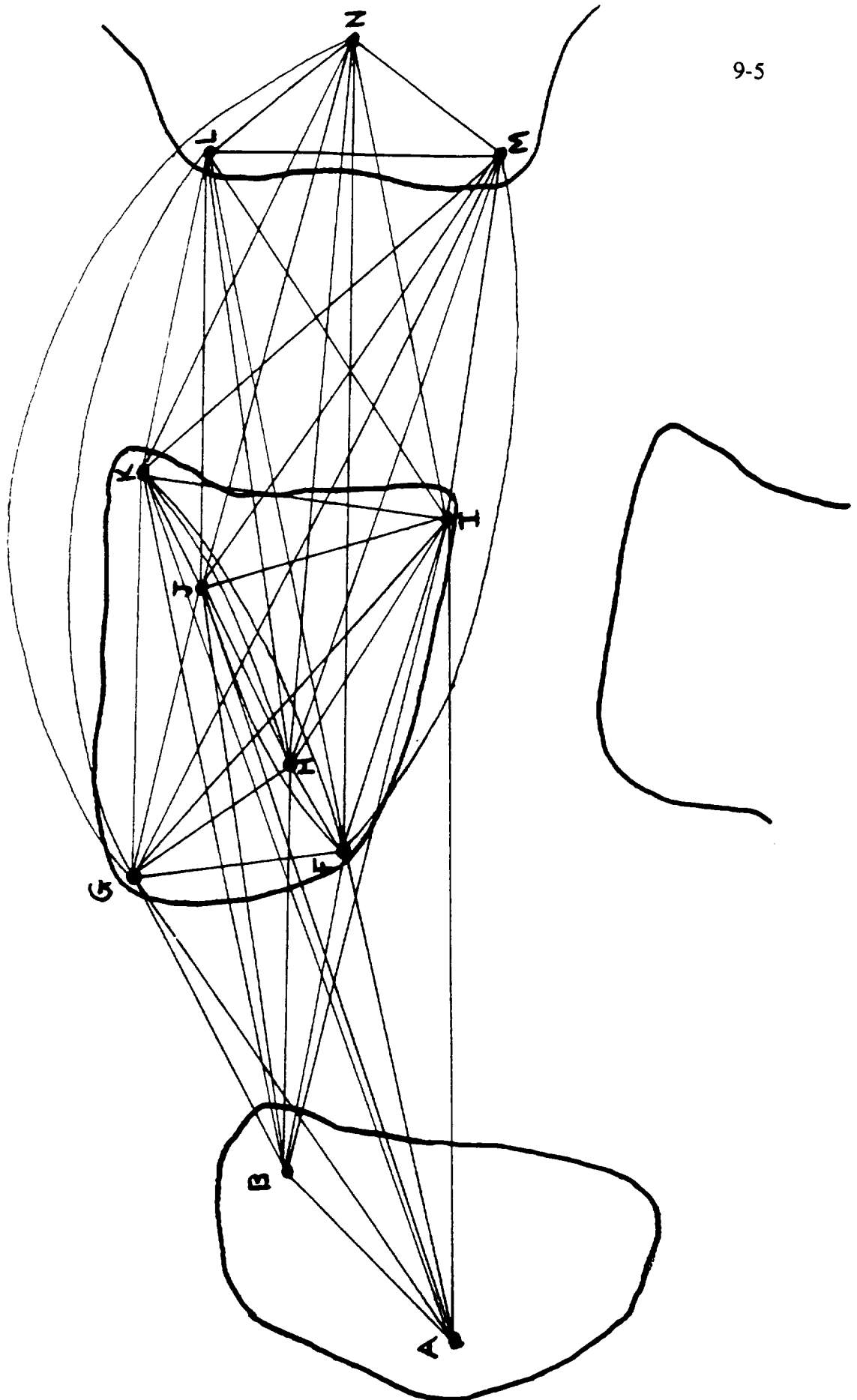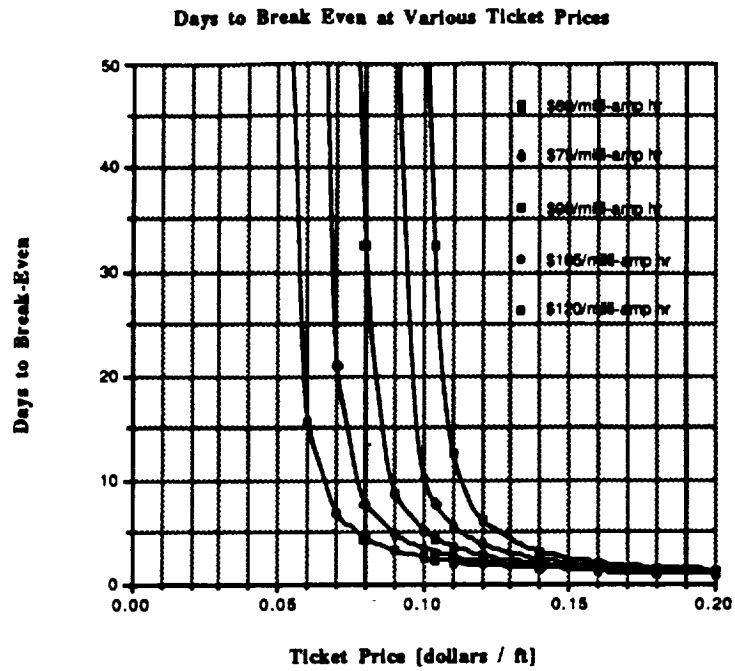
Figure 9.1 Designated Service Market

Days to Break Even at Various Ticket Prices

Figure 9.2 Days to Break Even for Various Ticket Prices
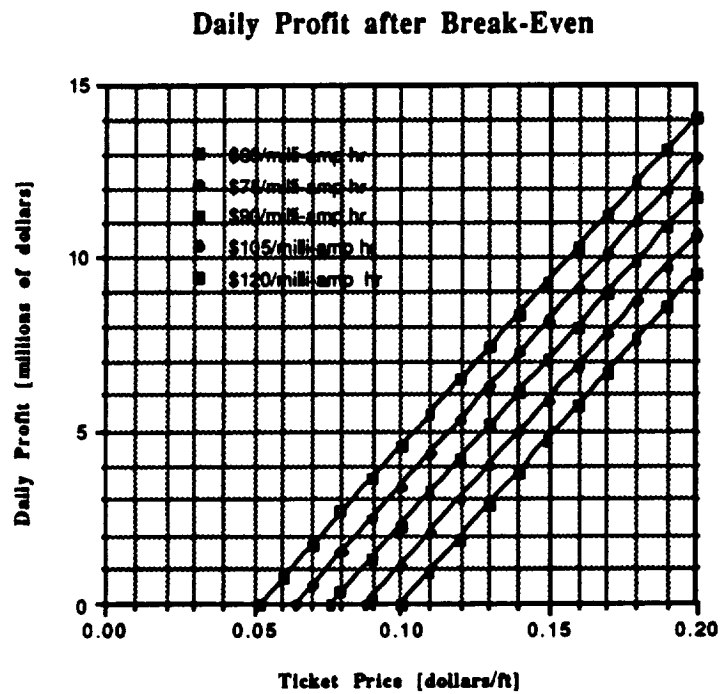
Daily Profit after Break-Even

Figure 9.3 Daily Profit After Expenses Are Recovered

10.0 DERIVATIVE AIRCRAFT

In many cases, a basic design of an aircraft is more than suitable for a given mission. However, there are times when an airline wants to expand its presence in a given market or try to enter another market but it simply does not have the funds to design and purchase a new aircraft. In cases such as these, a derivative aircraft is often the best solution.

A derivative aircraft builds upon the basic design and enhances the performance characteristics of that aircraft. Enhancements could include a lengthening of the fuselage, increase in wingspan or a change in the powerplant.

10.1   Fuselage Enhancements

A simple modification that can be made to the basic aircraft is to lengthen the fuselage. The benefits here are obvious. A larger fuselage can accommodate more passengers or cargo. This helps to increase profit per flight. But the penalty paid is in aircraft weight and range. A larger fuselage will also affect the moment arms for stability and control. Thus, the need for extra payload capacity needs to be carefully weighed against the penalties in performance.

10.2   Wing Enhancements

10.2.1   Wing Span Modifications

Another modification that can be made, albeit a major one, is a change in the wing geometry. In the case of the *Behemoth Apteryx*, performance characteristics are increased dramatically when the wingspan is increased. As in the case of the fuselage, the benefits are clear. A larger span increases the aspect ratio of the aircraft which in turn helps to decrease the effects of induced drag. In addition, the larger lifting surface decreases the stall speed of the aircraft and the thrust required for take-off. This helps to increase the efficiency of the aircraft.

Alpha Design feels that the wingspan can be increased to seven feet with no hinging. This is because a fleet of aircraft with five foot spans will already be in place, so a second fleet of aircraft servicing only seven foot gates would not pose any problems. In fact, this would serve to effectively service all the gates within a given airport.

10.2.2   Winglet Addition

The addition of winglets is a low-cost alternative to major wing modification that will also improve the performance of an aircraft. By helping the wing lift distribution to become more elliptical, the induced drag can be reduced and overall aircraft lifting performance can be enhanced. As a result, the overall efficiency of the wing will be increased.

10.3   Powerplant Modifications

A third way that performance could be enhanced is through modifications to the powerplant. By increasing the power output of the engine, take-off thrust is increased. In the same vein, a larger engine can also carry a larger payload. This is one of the least viable options however, because a larger engine will increase weight in two ways. The larger engine adds weight and a larger engine requires a larger battery pack to run efficiently which adds weight. The compromise between weight and range would not prove worthwhile.

11.0 TECHNOLOGY DEMONSTRATOR

11.1 Configurational Information

The Behemoth Apteryx final design information is as follows:
The wing has a SPICA airfoil section, a span of 60 in., a chord of 14 in., efficiency of .76, and aspect ration of 4.28. The ailerons are 18 in. by 1.25 in. and have a maximum deflection of ± 15°. The fuselage is 44 in. long, 7 in. wide in the passenger section, 5 in. high at the highest point and tapering to 3 in. in the rear. The horizontal tail is 24.5 in. by 6.5 in. and mounted at -3°. The elevator is 1.25 in. in chord and runs the length of the horizontal tail. The maximum elevator deflections are ± 28°. The vertical tail is 10 in high, 6 in. at the root, and 4 in. at the tip. The rudder is 10 in. high and 1.5 in. in chord and has a maximum deflection of ± 25°. The c.g. ranges from .28c to .45c. The landing gear is a tail dragger configuration mounted at approximately .15c. The overall weight of the aircraft is 61.0 oz.

11.2 Flight Test Plan

Our plan for our flight test will be to take the aircraft off, fly the figure eight pattern as many times as possible, and land as safely as possible. We will measure the velocity, takeoff and landing distance, range, endurance, glide angle, and turn radius as accurately as possible. We will also qualitatively measure the handling qualities.

11.3 Test Safety Considerations

The only real safety considerations are for the spectators and the students taking data, the pilot, and camera man. For this we will have a preflight checklist to insure all fastenings are fastened, all systems are working, and that the spectators are all behind the safety net.

11.4 Flight Test Results

To be included upon conclusion of flight tests.

11.5 Manufacturing and Cost Details

Our final cost of our plane was $211,400.

The breakdown of costs was as follows:

Cobalt Astro-15 geared motor	$44,800
4 channel radio, receiver, & 4 servos	$56,000
Speed Controller	$28,000
Propeller	$ 1,200
12 cell Nicad battery packs	$18,400
Monokote (3 rolls)	$ 8,000
Other materials (balsa wood, glue, landing gear)	$40,000
100 hours of labor	$10,000

APPENDIX A: SECTION SUMMARIES

1.0 Mission

8000 foot range (includes loiter, redirect time, and landing)
50 passenger capacity
Take Off and Landing Distance ≤ 50 feet
Cruise Speed = 32 fps
Absolute Maximum Speed = 52 fps
Turning Radius ≤ 60 feet

Aircraft Configuration

Fuselage:
- 7 inches wide at the widest point, tapering down to 5 inches
- Height is 5 inches, also tapering down to 2 inches
- Total Length = 44 inches

Passenger Compartment:
- 2 door access fore and aft $^{w/}$ lavatories across from each exit
- Pilot's station mounted fore of the passenger compartment
- 17 seating rows: 16 rows $^{w/}$ 3 seats, 1 row $^{w/}$ 2 seats

Propulsion:
- ASTRO 15 electric motor $^{w/}$ Topflight 10-4 propeller

Wing:
- 60 inch span, 14 inch chord
- 2° wing dihedral
- no taper, sweep, or twist

Control Systems:
- ailerons mounted on the outer edge of the wings
- rudder
- elevators

Structure:
- balsa wood frame
- high stress components made of plywood or spruce
- Aircraft skin made of monokote

2.0 Concept Selection

Concept #1:
- conventional aircraft $^{w/}$ top mounted wing
- wing taper $\approx$ .667
- AR = 6
- Wing span = 60 inches
- C_{root} = 12 inches, C_{tip} = 8 inches
- passenger capacity = 50
- twin vertical tails
- "Tail Dragger" landing gear configuration

Concept #2:
- conventional aircraft $^{w/}$ top mounted wing
- C_{root} = 12 inches, C_{tip} = 8 inches
- Wing span = 84 inches
- passenger capacity = 50
- "Tail Dragger" landing gear configuration

3.0 <u>Aerodynamics</u>

SPICA airfoil:
- Cl_{max} = 1.2, l/d_{max} = 67
- Main wing mounted at a 7° incidence angle
- Maine Wing has a 60 inch span, 14 inch chord
- Optimum operating range occurs at Re ≈ 200,000
- L/D_{max} for the plane = 10.8
- Wing is easy to construct due to flat bottomed airfoil

4.0 Stability and Control

$C_{m\alpha}$ for the wing = .004832/degree
$C_{m\alpha}$ for the fuselage =.0000402/degree
$C_{m\alpha}$ for the plane = -.0118/degree
$C_{l\delta\alpha}$ = .0011
Wing dihedral = 2°

Horizontal tail:
- mounted at -3°
- $C_{m\delta e}$ = .923 (this allows for a cruise elevator trim angle of 0°)
- elevator chord = 1.25 inch along span of horizontal tail
- Minimum surface area for horizontal tail = 49 in^2
- Actual size: 24.5 inches x 6.5 inches
- AR_{tail} = 3.77

Acceptable Range for c.g. = .28c ≤ c.g. ≤ .45c (optimum location = .33c)

Vertical Tail sizing:
- 6 inch root, 4 inch taper, 10 inch height
- $C_{n\beta}$ = -.0000185/degree
- Se/St = .40
- rudder sizing = 1.75 inch x 10 inch
- rudder effectiveness ≈ .0225/degree

Ailerons:
- 18 inches x 1.25 inches
- mounted on the outermost edge of the wings

5.0 Propulsion

Motor	Cobalt Astro-15
Motor weight (including mount)	10.24 oz.
Propeller	Top-Flight 10-4
Prop efficiency at cruise	0.74 (from Apple IIe program)
Prop rpm at cruise	5307
Estimated static thrust	2.6 lb (from database)
Cruise power setting	84%
Cruise range (steady, level flight)	15 455 ft
Battery capacity	650 mah
Battery pack voltage	14.4 (nominal) 16.2 (peak)
Battery pack weight	12.67 oz.

Model	Company	Capacity [amp-hrs]	Weight of each cell [oz]	Voltage per cell
650SCR	Gates	0.650	.92	1.2
P-90SCR	Panasonic	0.900	1.23	1.2
P-120SCR	Panasonic	1.200	1.66	1.2
N-600SCR	Sanyo	0.600	1.02	1.2
N-900SCR	Sanyo	0.900	1.38	1.2
N-1200SCR	Sanyo	1.200	1.84	1.2

V_{cruise} = 32 fps
Cdo = .04
Propeller = Topflight 10-4

6.0 Weight and Center of Gravity

Building Materials	E (psi)	density, ρ (lbf/in^3)	stress, σ_{xx} (psi)
Balsa Wood	65,000	.0058	400
Spruce Wood	1,380,000	.0160	6,200
Plywood	2,010,000	.0231	2,500

COMPONENT	WEIGHT oz	WEIGHT %	X-COORDINATE	Y-COORDINATE
ENGINE	8.76	14.21	2.00	2.50
BATTERY	12.67	20.54	6.00	3.00
WING	9.00	14.59	10.50	5.50
FUSELAGE	11.6	18.80	17.40	2.80
EMPENNAGE	4.15	6.73	44.50	4.00
MAIN GEAR	4.15	5.54	6.00	-2.50
REAR GEAR	0.5	.81	44.00	-1.50
AVIONICS	6.58	10.67	11.50	2.00
PROPELLER	0.50	.81	0.00	2.50
PAYLOAD	4.50	7.29	27.50	0.75

7.0 Structural Design

Ultimate load factor = 2.5
Limit load factor = 1.8
Maximum normal load factor (during banking turns) = 1.4
Maximum achievable speed = 52 fps (based on power available)

Wing:
- 60 inch span, 14 inch chord
- Main wing spar located at 30% chord
- No sweep, taper, or twist
- Constructed with balsa wood and monokote
- Outboard mounted ailerons

Fuselage:
- Simple truss structure
- Split-level: lower for passengers, upper for mechanicals
- Components made with balsa wood and spruce
- Motor mounts and landing gear constructed with aluminum
- Main landing gear: 2 inch foam wheels
- Rear landing gear: 1 inch plastic wheel

Empennage:
- Elevator measures 24.5 inches x 1.25 inches
- Rudder runs length of vertical tail (10 inches)

8.0 Performance

50 passenger capacity + 2 crewmen
Take off and Land ≤ 50 feet

Friction coefficient $\mu \approx .04$
Take off speed = 29 fps
Cruise altitude = 20 feet
Cruise speed = 32 fps (35 fps maximum)

data based on .600 milli-amp hours after take off	FLIGHT TIME (optimum conditions)	RANGE (optimum conditions)	RATE OF CLIMB
MINIMUM based on steady, level flight	8.20 minutes @ V = 25 fps	12306.7 feet @ V = 25 fps	5.72 fps @ V = 25 fps @ 14.4 Volts
MAXIMUM based on steady, level flight	8.47 minutes @ V = 31 fps	19705.7 feet @ V = 44 fps	6.18 fps @ V = 31 fps @ 14.4 Volts
CRUISE based on steady, level flight	8.46 minutes @ V = 32 fps	16236.8 feet @ V = 32 fps	6.17 fps @ V = 32 fps @ 14.4 Volts

Power Required at Cruise = 15.04 Watts (@ i= 4.17 amps)
Aircraft L/D_{max} = 10.8

Minimum Glide Angle, γ = 5.28°
Best Gliding Range, $X_{best\ range}$ = 270.25 feet

9.0 <u>Economics</u>
PRODUCTION COSTS

Fixed Costs

Astro Cobalt 15 geared motor	$44,800 / aircraft
4 channel radio, receiver, & 4 servos	$56,000 / aircraft
Speed Controller	$28,000 / aircraft
Propeller	$ 1,200 / aircraft
12 cell Nicad battery packs	$18,400 / aircraft
Monokote (3 rolls)	$ 8,000 / aircraft
Other materials (balsa wood, glue, landing gear)	$40,000 / aircraft
150 hours of labor	$15,000 / aircraft

Total Fixed Cost   **$211,400 / aircraft**

Delivery Price   **$325,000 / aircraft**

Operating & Maintenance Costs

Fuel	$60-$120 / milli-amp hour
Maintenance	$500 / flight
Crew of 2 / flight	$200 / flight

Operating & Maintenance Costs

Fuel	$60-$120 / milli-amp hour
Maintenance	$500 / flight
Crew of 2 / flight	$200 / flight

Each aircraft will be delivered to the airline at a price of $300,000 per aircraft. For a fleet of 38 passenger aircraft, the total delivery price is $11,400,000. Since the cost to produce 38 aircraft will be $7,273,200, Alpha Design will realize a profit of $4,126,800 on the sale of the aircraft.

APPENDIX B: TK!SOLVER CODES

Cruise 5/14, 10-4:

Variables

	Q	1.217536	psf	dynamic pressure
.002378	rho		slug/ft3	air density
32	vel		ft/sec	air speed
	Cd	.04882011		a/c drag coefficient
.02	Cdo			zero lift drag coefficient
	Cl	.54305081		a/c lift coefficient
.76	eff			efficiency factor
4.2857	AR			aspect ratio
1	n			load factor
3.8568841	W		lb	a/c weight
5.8333	S		ft-ft	wing area
	Preq	15.042637	W	a/c power required - level flight
	ROC	9.3757528	ft/s	rate of climb
	Pavail	64.068115	W	power available from propeller
	v	15.182774	volt	armature voltage
14.4	vset		volt	battery voltage
.1059	Kb			battery constant
	i	9.6055295	amp	motor current draw
	motrpm	17714.786	rpm	motor speed (rpm)
.12	Ra		ohm	armature resistance
.000792	Kv		volt/rpm	motor speed constant
	proprps	8015.7405	rpm	propeller speed (rps)
2.21	gr			gear ratio
	J	.28744595		propeller advance ratio

.8333	propd		ft	propeller diameter
	eta	.67571924		propeller efficiency
1.084	Kt		in-oz/amp	motor torque constant
.95	greff			gear efficiency
	fltime	3.1544331	min	flight time
.505	batcap		amp-hr	battery capacity
	range	6056.5115	ft	range
	CT	.07216236		
	CP	.03069733		

Rules

$Cd=Cdo+Cl^2/(PI()*eff*AR)$
$Cl=(n*W)/(Q*S)$
$Preq=Q*S*Cd*vel$
$ROC=(Pavail-Preq)/W$
$v=vset-Kb*i$
$motrpm=(v-i*Ra)/Kv$
$proprps=motrpm/(60*gr)$
$J=vel/(proprps*propd)$
$CT=Ct(J)$
$CP=Cp(J)$
$eta=Ct(J)*J/Cp(J)$
$Pavail=eta*Cp(J)*rho*proprps^3*propd^5$
$Cp(J)*rho*proprps^3*propd^5=((Kt/Kv)*(v*i-i*i*Ra)*.0005454-floss(motrpm))*greff$
$fltime=batcap/i$
$range=vel*fltime*3600$

Ground Roll:

	phi	.87671233	n/a	THESE ARE THE VALUES THAT WERE HELD CONSTANT AS EACH OF THE
.83333333	h		feet	INDIVIDUAL VARIABLES WERE
5	b		feet	ALTERED IN ORDER TO ASCERTAIN
	Lift	2.7214163	lbf	THEIR IMPACT ON THE GROUND
	Vel	29.038866	fps	ROLL AND THE THRUST.
	Vstall	24.199055		
	S	5.8333333	ft^2	
.95	Cl		n/a	
	Drag	.28175421	lbf	WHEN THE THRUST WAS BEING

.02	Cdo			MONITORED THE GROUND ROLL,
.75	effy			Xgr, WAS HELD AT THE MAXIMUM
	AR	4.2857143		ACCEPTABLE LIMIT OF 50 FEET
	Weight	3.8568825	lbf	AS DETERMINED BY THE
50	Xgr		feet	GROUP'S MISSION DEFINITION.
	Thrust	1.4382201	lbf	WHEN THE GROUND ROLL WAS BEING MONITORED, THE THRUST
.04	mu			WAS HELD CONSTANT AT 1.5 LBF,
1.1666667	chord		feet	WHICH WAS EVALUATED AS A GOOD
	TOLift	5.5539108		GUESS AT THE MAXIMUM
	TODrag	.57500859		PRODUCABLE THRUST.

Rules

$phi=((16*h/b)^2)/(1+(16*h/b)^2)$

$Lift=.5*.002377*(0.7*Vel)^2*Cl*S$

$Drag=.5*.002377*(0.7*Vel)^2*S*(Cdo+phi*Cl^2/(3.141592654*effy*AR))$

$Vel=1.2*SQRT(2*Weight/(.002377*S*Cl))$

$Xgr=1.10*(1.44*Weight^2/(32.2*.002377*S*Cl*(Thrust-(Drag+mu*(Weight-Lift)))))$

$S=b*chord$

$AR=b^2/S$

$TOLift=.5*.002377*Vel^2*Cl*S$

$TODrag=.5*.002377*Vel^2*S*(Cdo+phi*Cl^2/(3.141592654*effy*AR))$

$Vstall=Vel/1.2$

$Weight=.096429*S+3.29438$

Landing Roll:

Variables

	phi	.87671233		
.83333333	h		feet	
5	b		feet	
	Lift	3.1938844	lbf	
	Vel	31.458772	fps	
.95	Cl			
	S	5.8333333	ft^2	
	Drag	.32703277	lbf	
.02	Cdo			

.7605	effy			
	AR	4.2857143		
	Weight	3.8568825	lbf	
60	Xgr		feet	
	Thrust	1.8326546	lbf	
.04	mu			
1.1666667	chord		feet	
	Vstall	24.199055	fps	
	RThrust	.73306183	lbf	REVERSE THRUST(40% THRUST CAPACITY)

Rules
phi=((16*h/b)^2)/(1+(16*h/b)^2)
Lift=.5*.002377*(0.7*Vel)^2*Cl*S
Drag=.5*.002377*(0.7*Vel)^2*S*(Cdo+phi*Cl^2/(3.141592654*effy*AR))
Vel=1.3*SQRT(2*Weight/(.002377*S*Cl))
Xgr=1.10*(1.69*Weight^2/(32.2*.002377*S*Cl*(RThrust+(Drag+mu*(Weight-Lift)))))
S=b*chord
AR=b^2/S
Vstall=Vel/1.3
Weight=.096429*S+3.29438
RThrust =0.4*Thrust